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SHUTTLE

ALTERNATE SPACE SHUTTLE CONCEPTS STUDY

3 NOVEMBER REVIEW • NAS 8-26362

LOCKHEED MISSILES & SPACE COMPANY

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ALTERNATE
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3 NOVEMBER REVIEW

LOCKHEED MISSILES & SPACE COMPANY
SUNNYVALE CALIFORNIA

AGENDA

The chart opposite shows the NASA-furnished agenda for the 3 November 1971 review of the Space Shuttle Phase B Extension studies. Each of the subjects is treated in order in the following charts.

$\boxed{\text{INTRODUCTION}}$

ORBITER & TANKS

- H VS HO
- TANDEM VS PARALLEL
- INTERSTAGE CONCEPTS
- TANK SEPARATION & DISPOSAL
- TANK DESIGN/COST

MARK I/MARK II ENGINE BASELINE

- J-2 OR J-2S ⟶ HiP_C
- J-2 OR J-2S FOR MK I/MK II
- HiP_C FOR MK I/MK II
- ENGINE PROGRAM RECOMMENDATION

SUMMARY OF BOOSTER STUDIES

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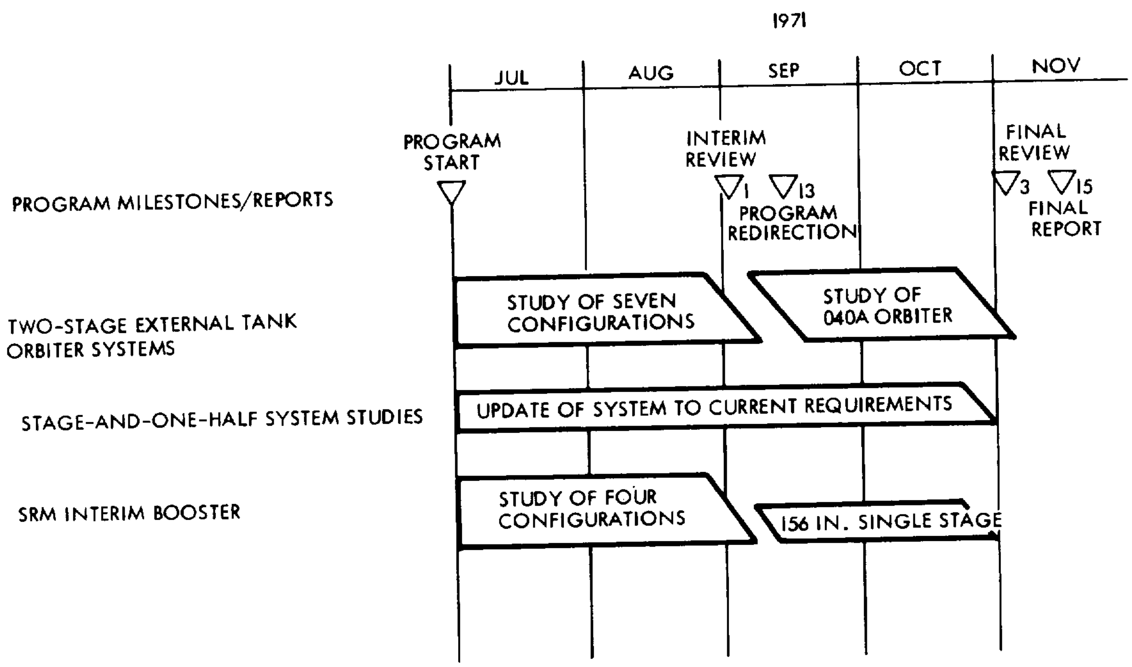
- AVIONICS
- TPS
- STRUCTURE (ALUMINUM VS TITANIUM)

D05786(1)

At the initiation of the four-month extension, Lockheed was to study two-stage external tank orbiter systems utilizing SRM interim boosters and the stage-and-one-half system. At the interim review on 1 September, results of the initial screening were presented along with Lockheed's recommendations.

In mid-September, NASA provided direction for the remainder of the study. The orbiter analysis work was concentrated on the O4OA system, an external tank delta-wing orbiter configuration launched on either a reusable LOX/RP booster or a reusable pressure-fed ballistic booster. Work was to continue at a low level on the stage-and-one-half system and the Lockheed-recommended SRM booster.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STUDY OF 040A ORBITER
STAGE-AND-ONE-HALF SYSTEM STUDIES
UPDATE OF SYSTEM TO CURRENT REQUIREMENTS
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The chart opposite shows two tandem launch configurations utilizing the O4OA-type orbiter and an RS-1C type heat-sink booster. The configuration on the left utilizes an external hydrogen tank and the one on the right an external hydrogen tank plus oxygen tank.

For the external hydrogen H tank system, the orbiter will be about 25-ft longer, thus increasing system weight and cost as shown. The smaller hydrogen tank decreases recurring costs, but the added complexity of having to size the orbiter to changing velocity requirements more than offsets this advantage.

H VS HO EXTERNAL TANKS

H ONLY

ADVANTAGES

- REQUIRES SMALLER TANK

  33 KLB VS 51 KLB

- PROVIDES LOWER RECURRING COST/FLIGHT

  DIFF. ≈ $400K/FLT

H + O

ADVANTAGES

- REQUIRES SMALLER ORBITER

  115 FT VS ≈ 140 FT

- DECOUPLES ORBITER FROM STAGING VELOCITY AND MISSION VELOCITY
- REDUCES PEAK ANNUAL FUNDING

  DIFFERENCE ≈ $45M
- REDUCES SYSTEM GLOW

  DIFFERENCE ≈ 350KLB
- REDUCES TOTAL PROGRAM COST

  DIFFERENCE ≈ $80M

H VS HO EXTERNAL TANK SELECTION

The choice of external HO tanks rather than H only tanks is recommended for the reasons shown.

H VS HO EXTERNAL TANK SELECTION

EXTERNAL HO IS RECOMMENDED, SINCE IT

- DECOUPLES ORBITER DEVELOPMENT FROM STAGING AND FROM TOTAL VELOCITY REQUIREMENT
- MINIMIZES ORBITER AND SYSTEM SIZE
- LOWERS PEAK ANNUAL FUNDING AND TOTAL PROGRAM COST

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PARALLEL VS TANDEM STAGING

One reason for recommending the tandem configuration earlier in the study was the greater orbiter-to-booster interface commonality it offered when changing from the interim SRM booster to the recoverable, heat-sink booster. In the current, 040A Mark I/ Mark II configuration, the same booster is assumed throughout the program, hence that advantage of the tandem arrangement does not apply.

However, as shown in the chart, significant advantages in the tandem arrangement tend to confirm the earlier conclusion. Since a design analysis was not conducted for the parallel 040A configuration, quantitative values cannot now be given for each factor, except for tank weight as shown on the subsequent chart.

PARALLEL VS TANDEM STAGING

PARALLEL

ADVANTAGES

- LOWER BOOSTER LOADS

DISADVANTAGES

- REQUIRES TWO SIDE-MOUNTED TANKS
- INCREASES ORBITER LOADS AND ORBITER WEIGHT
- MORE COMPLEX TANK STAGING
- MORE DIFFICULT BOOSTER/ORBITER ABORT SEPARATION BECAUSE OF BERNOULLI EFFECT
- HIGHER COST AND RISK SYSTEM

TANDEM

ADVANTAGES

- SINGLE LOWER COST TANK
- LIGHTER LOWER COST ORBITER
- SIMPLER TANK STAGING
- SIMPLER BOOSTER/ORBITER SEPARATION
- LOWER SYSTEM COST AND RISK

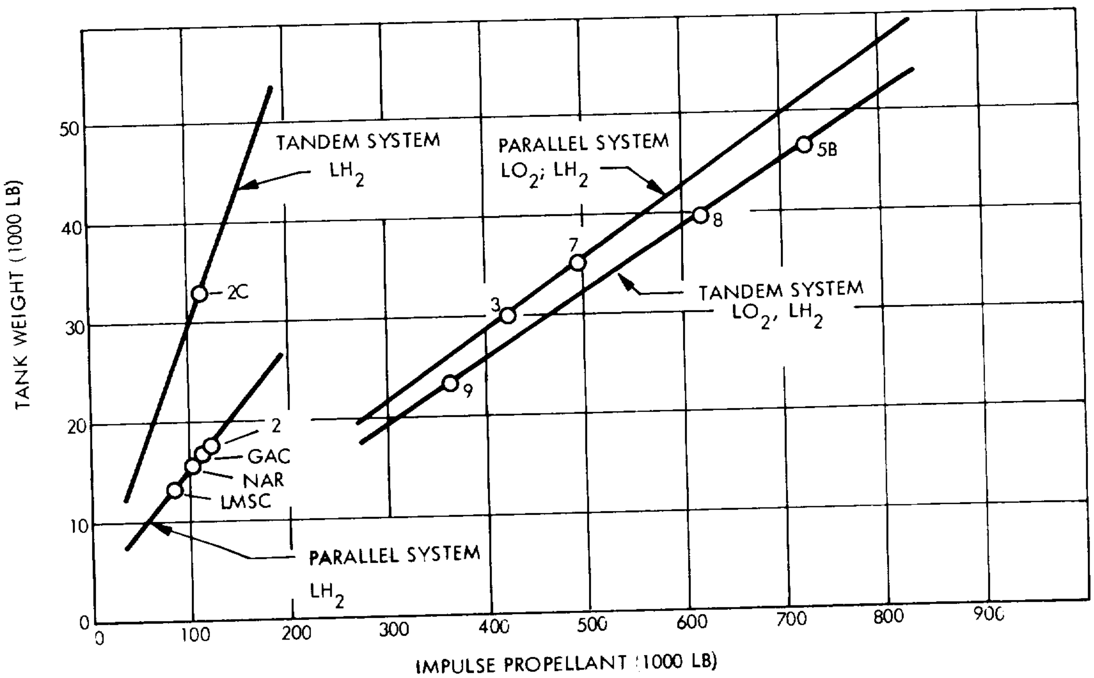
DISADVANTAGES

- MORE COMPLEX BOOSTER ATTACH ARRANGEMENT
- HIGHER BOOSTER LOADS

PARALLEL VS TANDEM: TANK WEIGHT COMPARISON

The tradeoff study of parallel vs tandem arrangement conducted in August was based on external hydrogen only rather than on external hydrogen plus oxygen. If the study had been based on external HO tank systems, the tank weight differences and consequently, the system weight differences and system cost differences would be decreased. This is evident from the chart opposite which shows differences of almost a factor of two in tank weights between tandem and parallel configurations using H tanks. For external HO tanks, the trend reverses, and the parallel configuration tanks are approximately 10 percent heavier than the tandem configuration tanks.

PARALLEL VS TANDEM: TANK WEIGHT COMPARISON



PARALLEL VS TANDEM SELECTION

The tandem arrangement is recommended for the O4OA system for the reasons shown. However, these are qualitative factors and need to be reduced to weights, costs, and technical risk comparisons wherever possible. Lockheed plans to redo this analysis under the follow-on study.

PARALLEL VS TANDEM SELECTION

TANDEM CONFIGURATION IS RECOMMENDED, SINCE IT

- SIMPLIFIES SEPARATION AT STAGING AND FOR ABORT
- REDUCES TANK WEIGHT AND COST ($\triangle$WT = 6000 LB; $\triangle$COST = \$120M)
- REDUCES ORBITER WEIGHT AND COST

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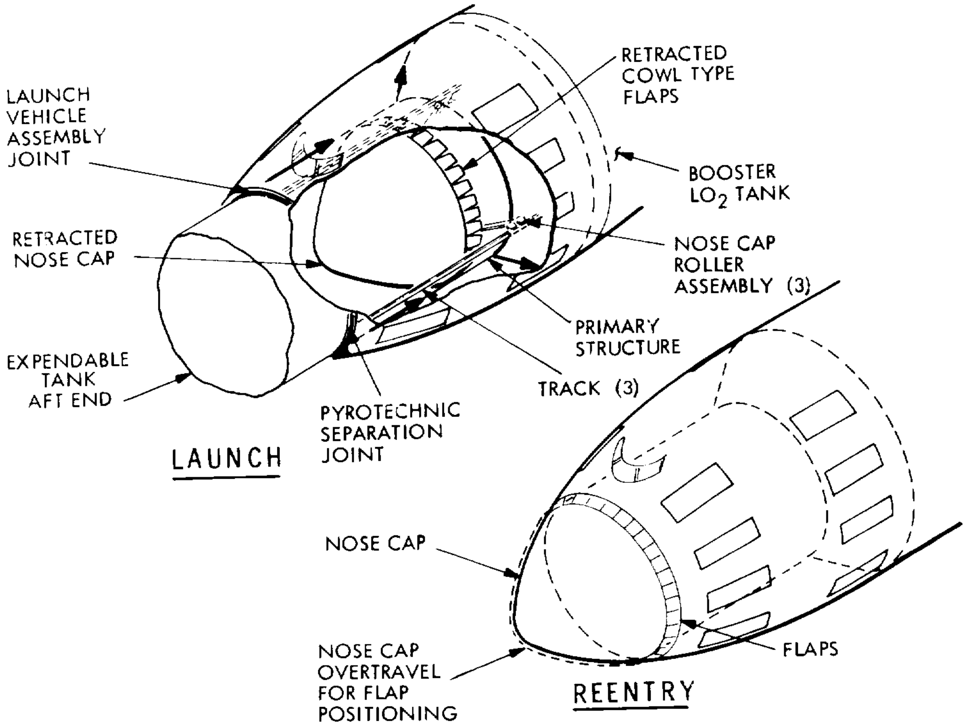
BOOSTER-ORBITER SEPARATION

(Retractable Nose Concept)

The retractable nose concept (baseline approach) permits the booster cabin, nose gear, and airbreather engine systems to be located in the booster nose section while retaining a simple, direct interstage structure (integral and recoverable with the booster).

In the concept illustrated, the orbiter expendable tank diameter is approximately 22 ft, and the booster tanks (integral with the fuselage) are 33 ft in diameter. Since these tanks and the interstage structure have a common centerline, an annular space with a maximum depth of approximately 6 ft, which is used for equipment packaging, is available in the nose section of the booster. The airbreathing engines along with their inlet and exhaust ducting are located within this space, resulting in no penetration of the booster nose section primary structure.

RETRACTABLE NOSE CONCEPT



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BOOSTER/ORBITER SEPARATION PETAL DOOR CONCEPT

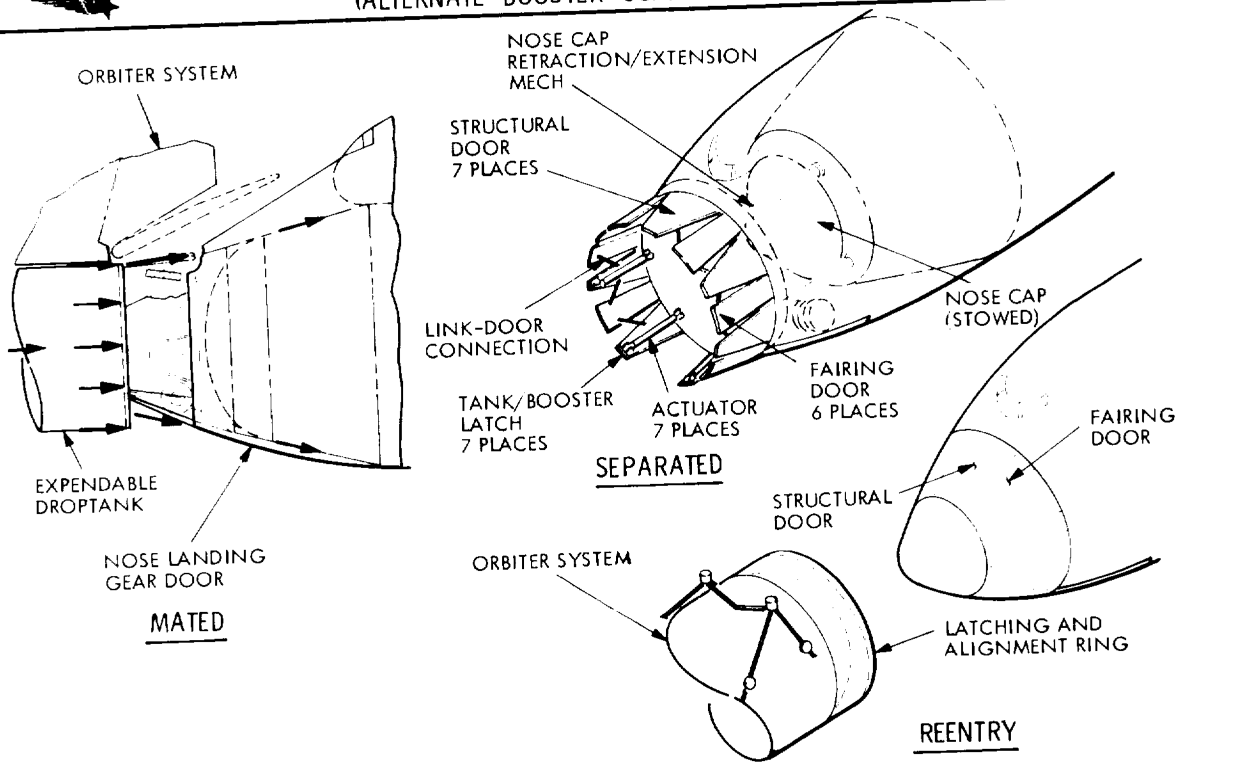
(Alternate Booster Configuration)

Recoverable booster arrangements, with their airbreather installations located in the after portion of the fuselage/wing system, have the greatest potential for a most direct and shortest load path between the booster noseload (orbiter system) and the heat-sink (thick shell) booster propellant tanks.

The interstage concept shown illustrates this potential arrangement. Booster propellant tanks can be brought as far forward in the arrangement as possible and, through the use of a system or structural/fairing doors, can be connected almost directly to the cylindrical portion of the aft end of the orbiter expendable tank.

This arrangement concept eliminates the requirement of discarding the mounting structure between the two-stage systems after their separation and minimizes the length of expendable tank aft skirt required and therefore the inert weight carried into orbit.

BOOSTER/ORBITER SEPARATION PETAL DOOR CONCEPT
(ALTERNATE BOOSTER CONFIGURATION)

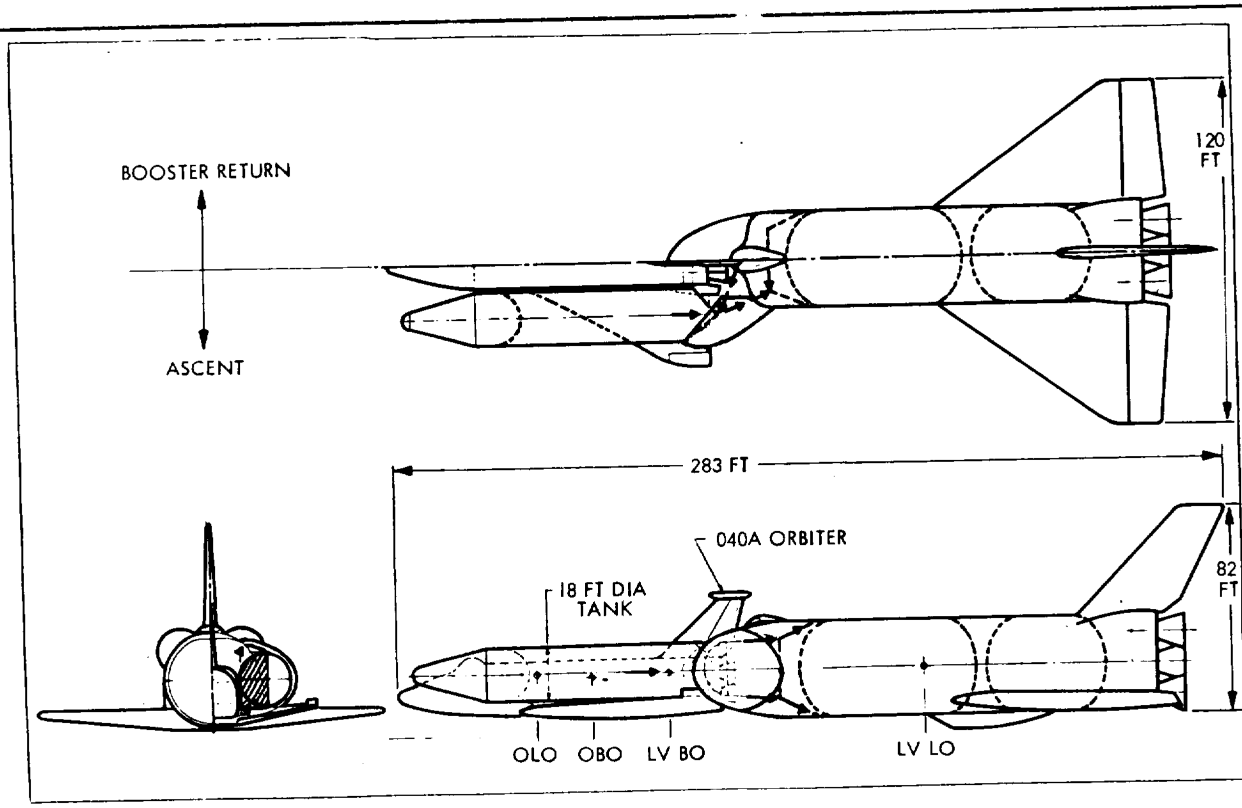


ALTERNATE BOOSTER/ORBITER INTERSTAGE CONCEPT

The launch vehicle concept shown illustrates an interstage approach associated with supporting a booster noseload consisting of a recoverable and reusable orbiter system, employing dual expendable HO tankage. The tank system has been rearranged from the baseline single tandem concept to relieve the eccentric loading of the orbiter element on the expendable tank/booster system during first-stage burn and on the tankage during second-stage operation. Rocket engine gimbal requirements are minimized for both propulsive stages.

To avoid the disposal problems associated with nonrecoverable interstage structures, a structural clamshell door system, as shown conceptually, can be used. In this system, the doors form the basic nose section of the recoverable booster. The booster noseload is attached to the open clamshell door at the aft end of each expendable tank. These attachment points also serve as the aft attach points for the expendable tanks-to-orbiter attachment system.

ALTERNATE BOOSTER/ORBITER INTERSTAGE CONCEPT



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TANK SEPARATION AND RETRO CONCEPTS

The tank separation concept is configuration dependent, i.e., a scheme which is desirable for a ventrally mounted single tank may be unacceptable for a pair of tanks mounted above the wings. The separation concept follows from the tank structural attachment method. To these attachments are added piston/cylinder separation devices, which are pressurized at 1000 psia to provide a separation force. The forward attachment can take 71,6000 lb limit load, and a system moment balance results in the aft attachments providing 7,000 lb of force each. The piston stroke is 24 in.; associated separation data are presented on the chart opposite.

The retro delta velocity is 300 fps, which is a compromise between lower values that require less retrorocket weight but result in substantially greater dispersions and higher velocities requiring a disproportionate increase in rocket weight compared to the resulting dispersions. Four rockets are peripherally mounted at the aft tank end and canted 10 deg so that, if one misfires, the remaining rockets will deliver 225 fps and assure a water-impact even though the dispersion may be greater than nominal.

TANK SEPARATION AND RETRO CONCEPTS

REQUIREMENTS

- INDIAN OCEAN NOMINAL IMPACT
- RETRO $\Delta V \simeq 300$ FPS
- TANK SEPARATION TRANSLATION $\simeq$ ONE-HALF TANK LENGTH
- DUMP RESIDUAL LIQUID AND VENT ULLAGE GAS

SEPARATION CONCEPT

- DUMP RESIDUAL LH_2 AND VENT
- YAW ORBITER: $\beta = 180^0$; $\alpha = 0$
- THREE POINT SEPARATION PER ATTACHMENT
- PRESSURIZED CYLINDER/PISTON SEP.
- SEPARATION DATA:

W_T	= 50K	t_{SEP}	= 0.26 SEC
F_{SEP}	= 91.6K	V_{SEP}	= 15.3 FPS
n_{SEP}	= 1.83g	t_{TRANS}	= 4.5 SEC

RETRO CONCEPT

- FOUR SRM RETROROCKETS
- RETROROCKET DATA

I_{S_V}	= 290 SEC
I_{T_V}	= 448K LB-SEC
W_P	= 400 LB (EA)
CANT	= 10 DEG

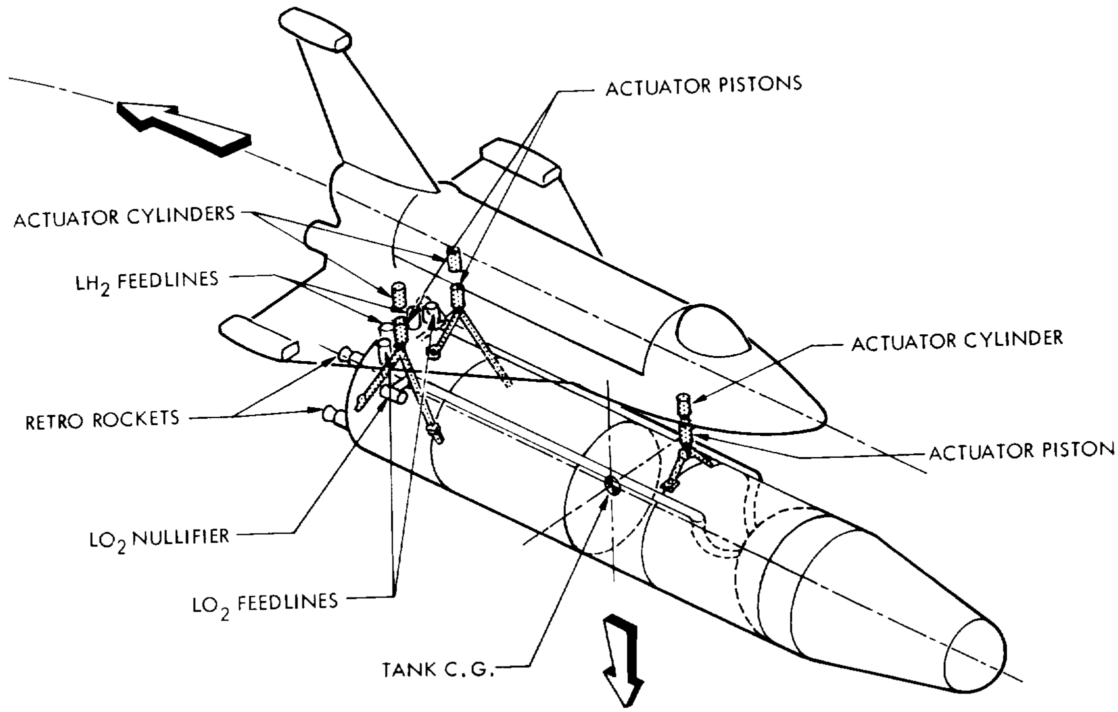
D05610

TANK/ORBITER ATTACHMENT CONCEPT

The external tank is attached to the orbiter at three points: one forward and two aft. This means that there will be three structural separations. Integral with the structure/mechanical attachment are three pistons that fit into cylinders, which after separation are pressurized to 1000 psia by a gas generator. The pressure acting on the cylinders provides a 1.8g acceleration for about a quarter of a second, resulting in a separation velocity of 15 fps.

Trade studies will be conducted for comparing the weight of this concept against one using the ACPS to separate the orbiter from the tank. The latter concept would result in a much lower translational velocity, which means that the resulting rate errors imparted to the tank would have to be quite low so that the effect on dispersion would be within acceptable limits.

TANK/ORBITER ATTACHMENT CONCEPT



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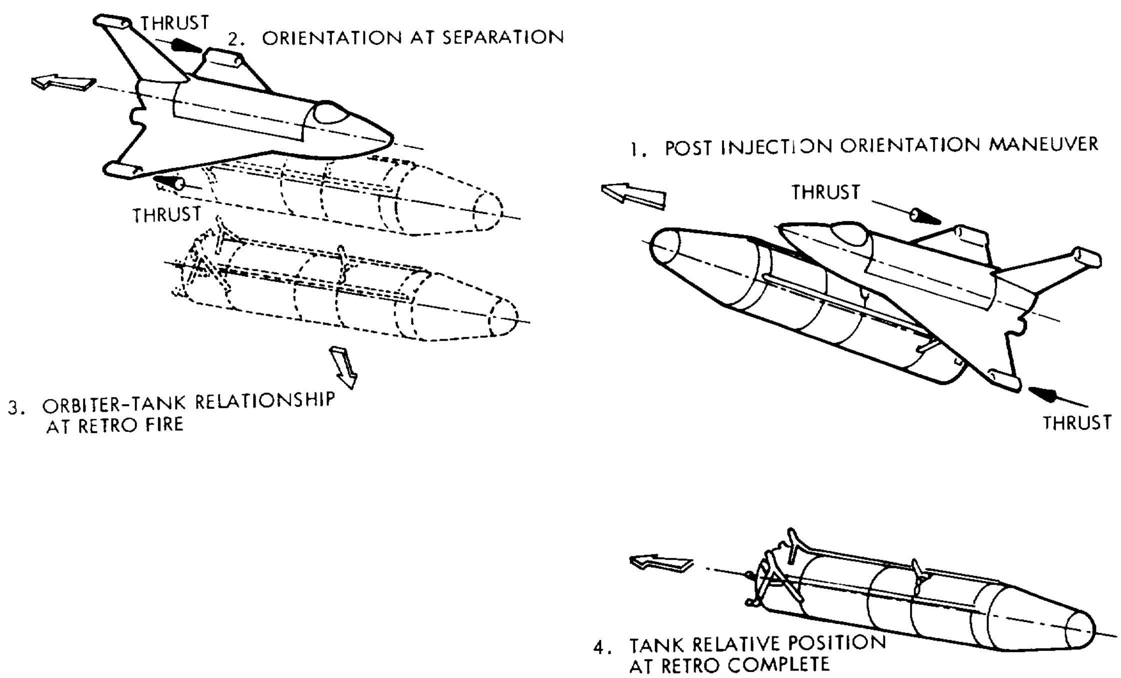
ORBITER-TANK SEPARATION AND RETRO SEQUENCE

For a nominal ascent, orbit injection will occur at an altitude of 50 nm to be followed by a nonpropulsive coast to 100 nm. It is during this coast period that various operations must be performed, including a 180-deg yaw maneuver, to facilitate tank separation and entry. While the vehicle is rotating, the tank vent operations will be completed. The hydrogen tank vent occurs first, because it is necessary to eliminate the liquid residual to reduce cg deviations, but also, depending on the insulation concept, the tank wall temperature may increase during coast which will reduce the structural allowable and cause a hazardous condition if the ullage pressure is not reduced. This reduction in pressure presumably will allow the tanks to enter intact to relatively low altitudes. Under such conditions, the dispersions should be within the bounds of intact entry, even when the possibility of trimmed lift of fragments is considered.

With the vehicle oriented in the proper attitude, the separation will be signalled, and the tank will be released and translated away from the orbiter. An adequate (safe) translation distance is not determined, but one-half tank length (about 60-70 ft) is suggested as a rule of thumb. A timed retro can be used to fire the retrorockets.

ORBITER – TANK SEPARATION AND RETRO SEQUENCE



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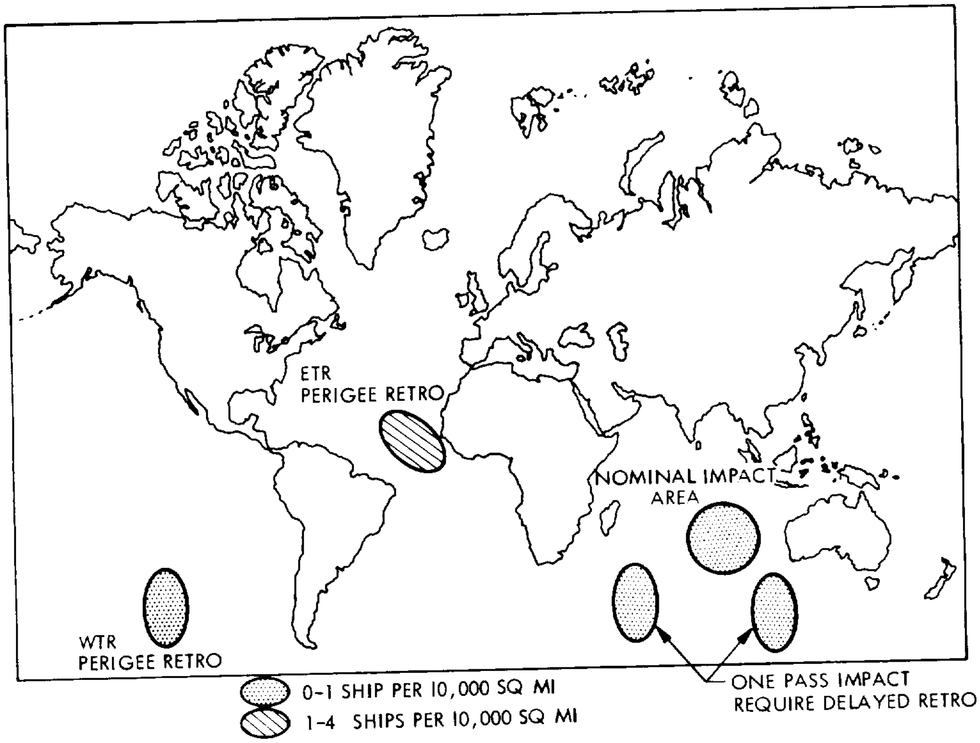
EXTERNAL TANK POTENTIAL IMPACT AREAS

The impact area of primary interest is the Indian Ocean. Here there are areas of very low-traffic density, and a common impact location can be targeted for all NASA missions out of KSC. This area is indicated by the circle on the map adjacent to and west of Australia. By properly timing the tank separation, its attitude at separation, and the retro ΔV, one can achieve an impact within the approximate bounds indicated by the circle. The actual intrack dispersions will be of the order of 1200-nm and one to two hundred miles crossrange. However, this will depend upon the tank mass property characteristics and the attendant system deviations, such as result from orbit, separation and retro errors. Atmospheric deviations and aerodynamics have a relatively small effect on range dispersions.

Alternative possibilities for tank impact location would be the Atlantic Ocean, but there are probable restrictions associated with this. For example, north-easterly launches out of KSC could involve risk of impacting land masses unless the system characteristics and deviations are well understood and controlled. Easterly launches are probably alright for water-impact west of Africa.

Southerly launches out of WTR afford the options of either impacting in the Pacific Ocean west of South America or in the Indian Ocean south of Madagascar. On the other hand, northerly launches out of either base can be accommodated by carrying the tank to orbit and delaying separation until in a position to accomplish a south Indian Ocean impact. There is the possibility for an Arctic Ocean impact, but there is also the danger of an impact in northern Canada, Greenland, or Asia.

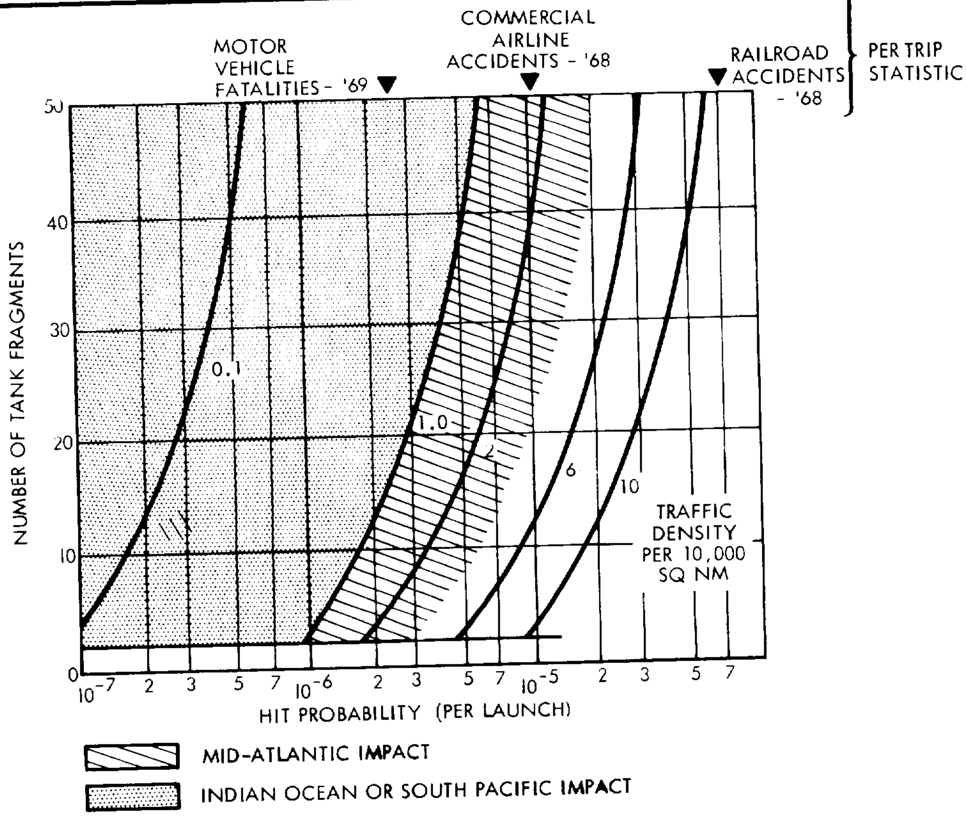
POTENTIAL IMPACT AREAS



HIT PROBABILITY VS TANK FRAGMENTS AND TRAFFIC DENSITY

The chart opposite shows the magnitude of risk involved in impacting shuttle tanks into various ocean areas. Shipping and aircraft densities in these areas are less than 0.1 craft per 10,000 square nm for the South Pacific impact area, from 0.1 to 1 craft per 10,000 square nm for the Indian Ocean impact area, and from 1 to 4 craft per 10,000 square nm for the mid-Atlantic impact area. Using an "effective" ship area, based upon a 645-ft long by 85-ft wide tanker, and assuming up to 50 major tank fragments, the highest hit probability for the preferred impact sites (South Pacific and Indian Ocean) is 6×10^{-5}, which is well below the present acceptable limit for safe launches.

HIT PROBABILITY VS TANK FRAGMENTS AND TRAFFIC DENSITY



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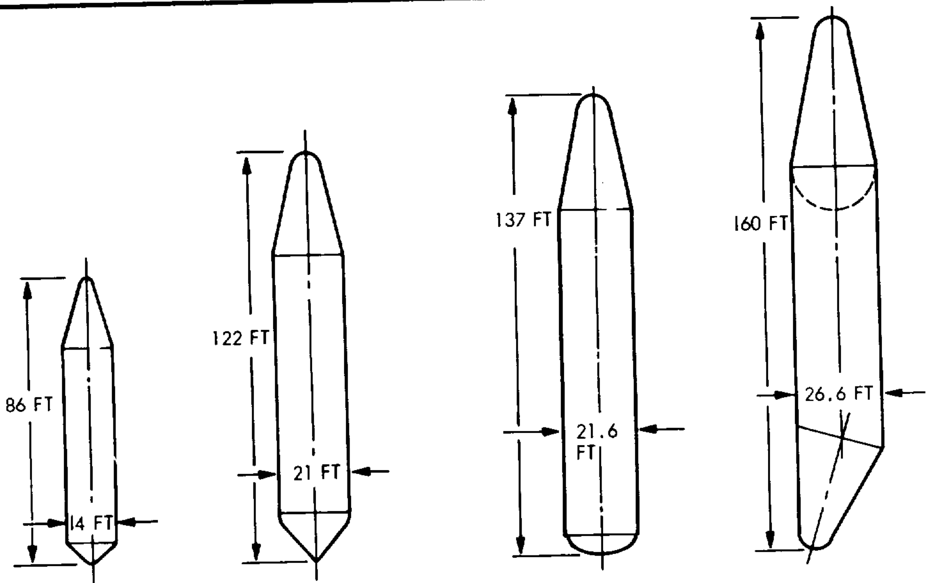
D05786(6)

The chart opposite illustrates the range of tank sizes that have been considered under the current ACS Contract. The baseline Task 4 tank is an LH_2 tank and is based on data furnished by GAC. The LO_2/LH_2 tank for configuration 5B carries boost loads and is designed to meet phased program requirements, i.e., to interface with both interim and final boosters. The O4OA LO_2/LH_2 tank is designed for the current two-and-one-half stage tandem baseline and is common to both Mark I and Mark II systems. The stage-and-one-half tank is the largest tank which has been considered and is the result of detail design studies extending over a period of three years.

EXTERNAL TANK SIZE RANGE



CONFIG.	BASELINE TASK 4	5B	040A	STAGE-AND-ONE-HALF
PROP., LB	42,250 (H)	787,000 (HO)	882,059 (HO)	1,533,000 (HO)
WEIGHT, LB	17,140	50,400	51,079	59,500

D05861

COST REDUCTION THROUGH WELD-BONDING

The chart opposite shows the cost reduction possible through the use of weld-bonding as compared to fusion-welding. Two tank configurations are represented: the Task 4 LH_2 tank (GAC baseline) and the stage-and-one-half exterior tank. A tank recurring-cost reduction of 10.3 and 25 percent, respectively, is achieved through the use of weld-bonding. Cost savings are primarily the result of savings in manufacturing and raw materials. The more complex tank (stiffened structure) makes possible a higher percentage of cost reduction.

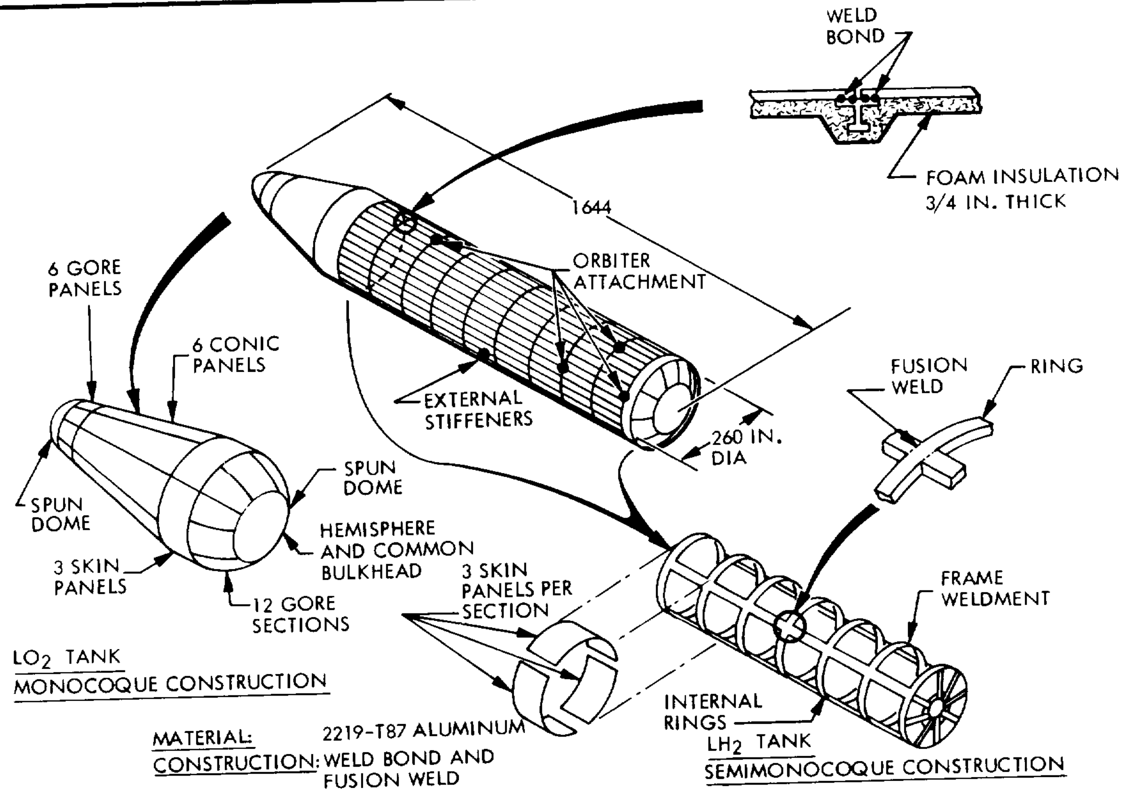
COST REDUCTION THROUGH WELD-BONDING

($ MILLIONS)

ITEM	BASELINE LH_2 TANK TASK 4 (MONOCOQUE)				STAGE-AND-ONE-HALF TANK (STIFFENED STRUCTURE)			
	FUSION WELDED	WELD BONDED	CHANGE ($ MILLION)	CHANGE PERCENT	FUSION WELDED	WELD BONDED	CHANGE ($ MILLION)	CHANGE PERCENT
RECURRING COST	650	583	67	10.3	1,621	1,209	412	25
MANUFACTURE	423	367	56	13.2	536	441	95	18
RAW MATERIAL	67	62	5	7.5	425	150	275	65

Current O4OA external tank weights and costs are based on the tank design shown. The forward LOX tank is of monocoque construction and is of fusion-welded aluminum. The LH_2 tank is of semimonocoque construction with external longitudinal stiffeners. A frame weldment composed of rings and longitudinal members is employed internally for additional stiffening. Skin panels are attached by weld-bonding. External insulation is used on the LH_2 tank.

EXTERNAL LOX/HYDROGEN TANK DESIGN



EXTERNAL TANK WEIGHT – O4OA ORBITER

The tank weights shown on the chart opposite reflect the current O4OA tank design. The LOX tank is of monocoque construction; the LH_2 tank, of semimonocoque construction. A combination of fusion-welding and weld-bonding is used in their fabrication. Internal insulation is used on the LH_2 tank only. A 2 percent contingency on dry weight is used.

EXTERNAL TANK WEIGHT – 040A ORBITER

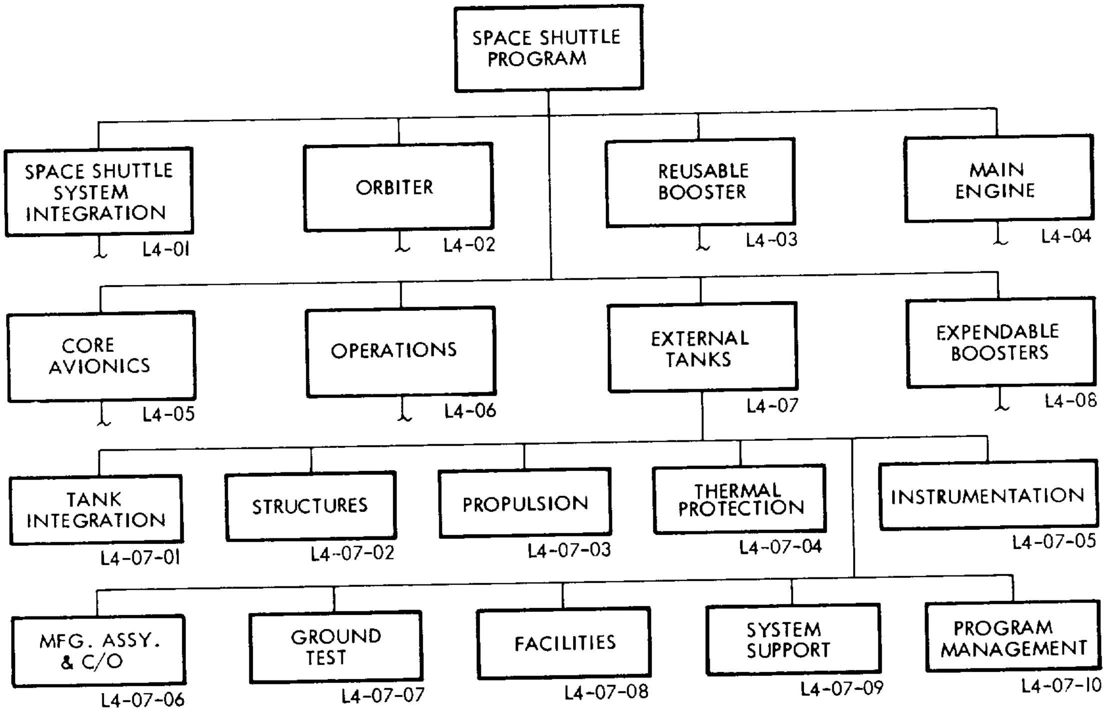
	LO_2	LH_2	TOTAL
MEMBRANE	7,933	15,387	23,320
STIFFENERS			
RINGS	225	1,664	1,889
LONGITUDINAL	0	10,818	10,818
TIE RINGS	0	3,845	3,845
ATTACHED STRUCTURE	0	350	350
STRUCTURE	(8,158)	(32,064)	(40,222)
INSULATION			3,480
PLUMBING			2,875
INST POWER			300
RETRO			3,200
CONTINGENCY (2 PERCENT)			1,002
EXTERNAL TANK TOTAL			(51,079) LB

λ' DRY = 0.945
TANK FACTOR = 0.0579
WEIGHT PROP = 882,059

D05669

WORK BREAKDOWN STRUCTURE – EXTERNAL TANKS

The work breakdown structure shown illustrates the primary cost elements that contribute to the total cost of the external tanks. LMSC has given special attention to this WBS element of the program since it is a significant portion of shuttle operating costs. Subelements are identified to facilitate accurate costing of this portion of the program and to aid in identification of cost drivers, as a means of tank cost reduction. Lockheed feels that cost control and reduction must be based on a complete listing and understanding of all tank cost elements.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PROGRAM MANAGEMENT
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EXTERNAL TANK COSTS – O4OA ORBITER

The chart opposite summarizes tank costs for the O4OA orbiter external tank system. The costs are based on the WBS elements previously shown, and reflect a combined fusion-welded and weld-bonded tank. Based on a learning rate of 89 percent and a total of 445 flights, total recurring costs are $457 million or $21.50/lb for the 51,000-lb tank.

EXTERNAL TANK COSTS - 040A ORBITER

DESIGN AND DEVELOPMENT	$134
TEST HARDWARE	16
MANUFACTURING FACILITY	30
TOTAL DDT&E	$180
RECURRING TANK SYSTEM COSTS	457
TOTAL	$637 *

FIRST-UNIT COST = $2.39M = 50 $/LB
AVG RECURRING COST = 21.50 $/LB
LEARNING RATE = 89 PERCENT

*445 FLIGHTS

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MARK I/MARK II ENGINE BASELINE

- J-2 OR J-2S $\longrightarrow$ HiP_C
- J-2 OR J-2S FOR MK I/MK II
- HiP_C FOR MK I/MK II
- ENGINE PROGRAM RECOMMENDATION

SUMMARY OF BOOSTER STUDIES

MARK I/MARK II SUBSYSTEM STATUS

- AVIONICS
- TPS
- STRUCTURE (ALUMINUM VS TITANIUM)

MAIN PROPULSION OPTIONS

The four main propulsion program options are shown. Two of the options provide engine changes between Mark I and Mark II and two options employ a common engine. The groundrule that the Mark II requirements size the external tanks was employed. It is important to note that both the Mark I and Mark II have the same size external tanks.

MAIN PROPULSION OPTIONS

MARK I		MARK II	EXTERNAL TANK* SIZING BASED ON 65K PAYLOAD DUE EAST
J-2 (TOTAL TWO VERSIONS)	→	Hi P_c	Hi P_c MARK II
J-2S (TOTAL SIX VERSIONS)	→	Hi P_c	Hi P_c MARK II
J-2S (TOTAL SIX VERSIONS)	SAME ENGINE →	J-2S (TOTAL SIX VERSIONS)	MARK II
Hi P_c	SAME ENGINE →	Hi P_c	MARK II

*MARK I AND MARK II HAVE SAME EXTERIOR TANK

D05691
llm

CRITERIA FOR RECOMMENDATION OF ENGINE PROGRAM APPROACHES

The criteria employed in examining the engine programs are indicated. Detailed evaluations were performed to provide the necessary information for applying the criteria.

CRITERIA FOR RECOMMENDATION OF ENGINE PROGRAM APPROACHES

PERFORMANCE CAPABILITIES

- MARK II 65K DUE EAST
- MARK I MINIMUM 10K POLAR, 25K DESIRED

COST ASPECTS

- TOTAL PROGRAM COST
- PEAK ANNUAL FUNDING

RISK CONSIDERATIONS

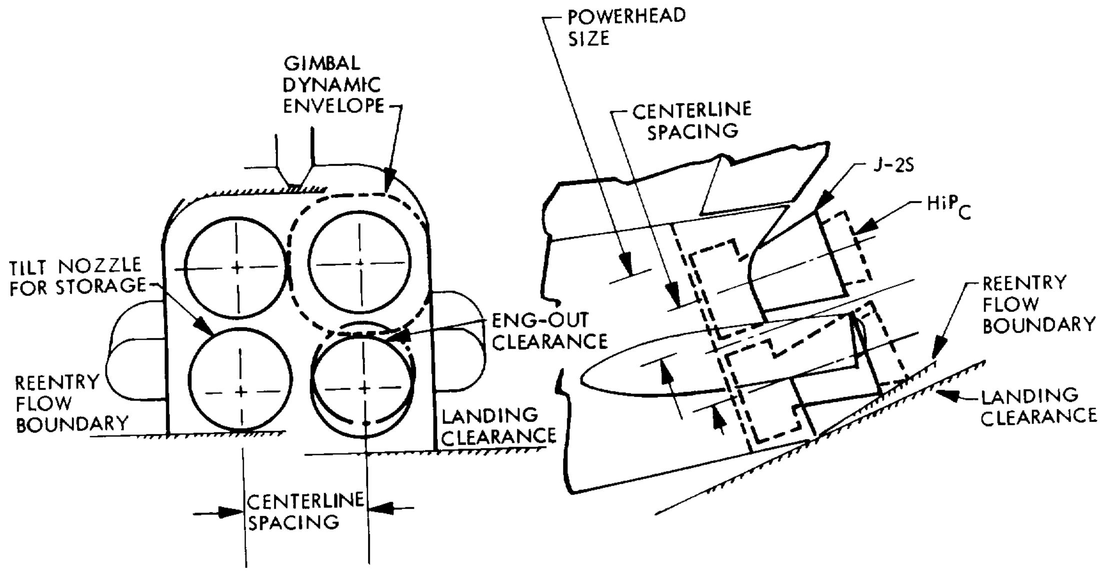
ABORT-TO-ORBIT CAPABILITY (NOT A REQUIREMENT)

D05692
llm

INFLUENCE OF ENGINE ENVELOPE ON INSTALLATION

Visualization of the engine installation envelope is aided by the opposite chart that shows the orbiter base region and the aft view, which is taken normal to the common thrustline of the four rocket engines. Extension of the gimbal dynamic envelope beyond the orbiter-fuselage moldlines is not a limiting condition, since canted nozzle operation is acceptable where cant angles are not large. The characteristic HiP_C rocket engine length is greater than its J-2/J-2S counterpart at the same thrust level and has a smaller diameter. Thus, a HiP_C rocket engine, which is sized to match the thrust level of a J-2/J-2S rocket engine, can fit within boundaries established by landing clearance and reentry flow fields.

INFLUENCE OF ENGINE ENVELOPE ON INSTALLATION



DO5909

RESULTING OPTIMUM OPERATING CONDITIONS

The chart opposite presents the orbiter propellant supply system for the four main engines. Each propellant is transferred by means of two main feedlines. The system is intended to accommodate both Mark I and Mark II engines without modifications.

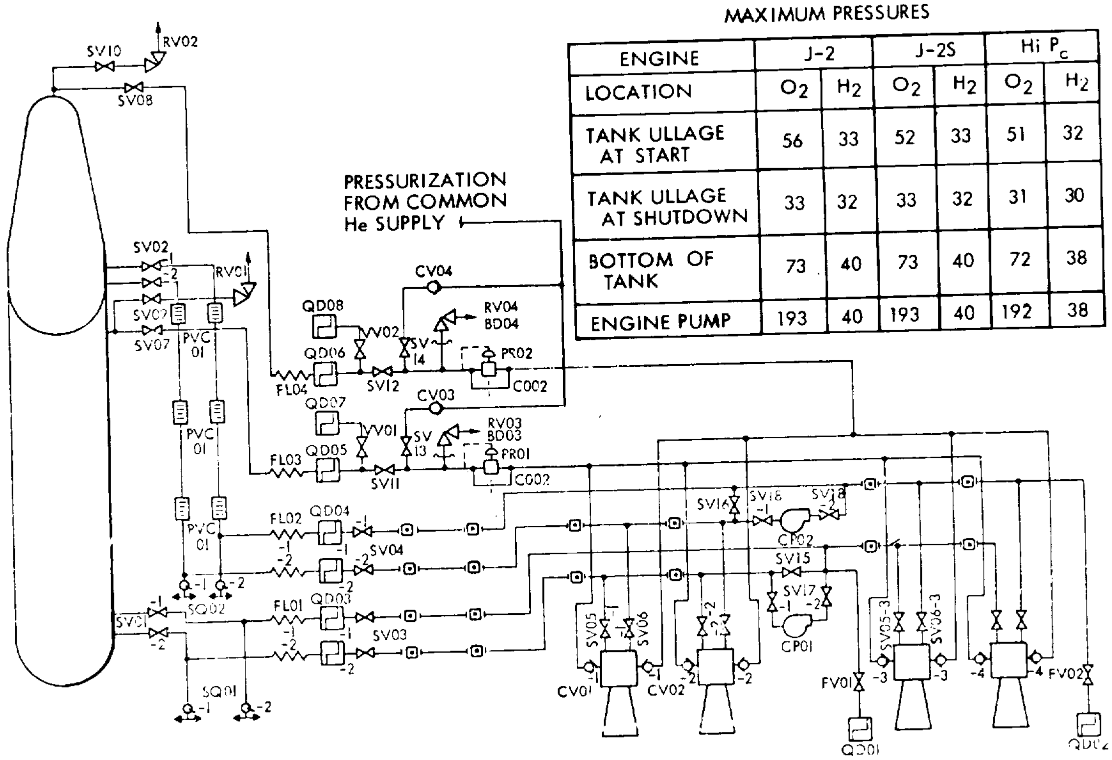
It is important to note that the required tank pressures for each of the three engines are optimized at approximately the same values. This allows the same tank design for an engine change between Mark I and Mark II without pressure-related penalties on either system. The feedlines were sized at 11.5 in. and the overall system weight is relatively insensitive to changes in the feedline diameters over a range of several inches.

Note that the high pressure shown for the oxygen system at the bottom of the tank and at the engine pump inlet are the result of the high hydrostatic pressures developed at maximum accelerations. The low-density hydrogen produces much lower hydrostatic pressures even though the liquid height in the tank is much greater.

RESULTING OPTIMUM OPERATING CONDITIONS

MAXIMUM PRESSURES

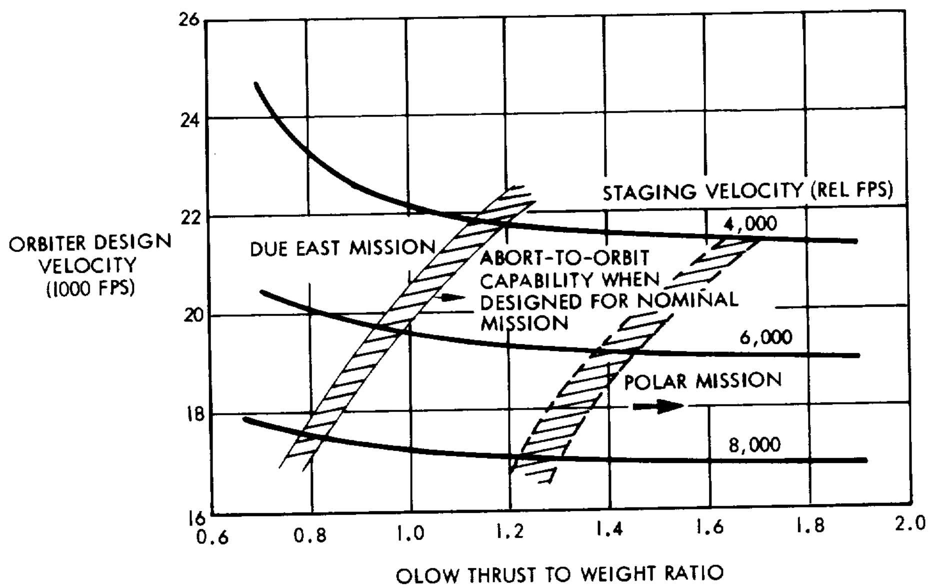
ENGINE	J-2		J-2S		Hi P_c	
LOCATION	O_2	H_2	O_2	H_2	O_2	H_2
TANK ULLAGE AT START	56	33	52	33	51	32
TANK ULLAGE AT SHUTDOWN	33	32	33	32	31	30
BOTTOM OF TANK	73	40	73	40	72	38
ENGINE PUMP	193	40	193	40	192	38



STAGING TO INJECTION REQUIREMENTS

The most significant parameters to orbiter design velocity are orbiter thrust-to-weight ratio and staging velocity. As either of these parameters decreases, the design velocity increases and importantly the rate of change in design velocity to these parameters also increases with the result that system sensitivities become higher.

The data shown below reflect ascent into a 50 x 100 nm due east orbit. Omitted for clarity are data for polar orbits for which a similar family exists. For reference purposes, the minimum orbit thrust-to-weight ratios, where abort to orbit can be attained without redesign if an engine is lost at staging, are indicated. The polar mission, by virtue of higher losses encountered for engine out and its lower on-orbit requirement (650 fps as compared to the due east of 1000 fps), becomes the critical mission requirement should abort to orbit be desired.



ORBITER DESIGN VELOCITY (1000 FPS)
26
24
22
20
18
16
0.6
0.8
1.0
1.2
1.4
1.6
1.8
2.0
OLOW THRUST TO WEIGHT RATIO
STAGING VELOCITY (REL FPS)
4,000
6,000
8,000
DUE EAST MISSION
ABORT-TO-ORBIT CAPABILITY WHEN DESIGNED FOR NOMINAL MISSION
POLAR MISSION

D05711
llm

OPTION

The first option examined was the program employing versions of the J-2 or J-2S engines for the Mark I with a change to the high-pressure engine on the Mark II. System sizing was based upon Mark II requirements.

OPTION

J-2 OR J-2S $\longrightarrow$ Hi P_c

DESCRIPTION AND GROUPING BY SIZE OF J-2 AND J-2S ENGINES

The importance of the available orbiter base area has been presented previously. The versions of the J-2 and J-2S engines, as defined by Rocketdyne - Division of North American Rockwell, have been grouped by nozzle diameter and engine length to provide a guide as to impact upon orbiter available base area.

The centerline spacings shown are for 7-1/2-deg gimbal angle and no canting of engines.

DESCRIPTION AND GROUPING BY SIZE OF J-2 AND J-2S ENGINES

ROCKET ENGINE	OPTION	F_{VAC}	I_S (NOM) (SEC)	CHAMBER PRESSURE (PSIA)	NOZZLE DIA (IN.)	ENGINE LENGTH (IN.)	CENTERLINE SPACING* (IN.)	ENGINE WEIGHT (LB)
J-2 J-2S J-2S	BASIC BASIC B-1	230K 265K 320K	425 436 434.5	780 1250 1520	80	119.5	96	3454 3800 4120
J-2	1	232.2K	429	780	91	143.5	111	3744
J-2S J-2S	A-1 B-2	272.5K 327.5K	448.3 446.8	1250 1520	112	175.5	136	3755 4040
J-2S J-2S	A-2 B-3	275K 330K	452.8 451.4	1250 1520	128	199.5	155	3855 4200

*NO CANT; 7-1/2-DEG GIMBAL

llm
DO5821

DESCRIPTION AND GROUPING OF ENGINES BY CHARACTERISTICS

The J-2 and J-2S versions and two thrust levels of the high-pressure engines have been grouped by characteristics. The changes shown to the J-2 or J-2S engines, respectively, allow accomplishment of the indicated thrust and specific impulse. The engines in the shaded regions do not satisfy spacing constraints imposed by the base area of the 040A vehicle. (Refer to previous chart).

The engine development costs to bring the engines up to Mark I status were obtained from data supplied by NASA/MSFC. It is known that these estimates do not include increasing the allowable inlet pressure requirements, which is discussed later.

DESCRIPTION AND GROUPING OF ENGINES BY CHARACTERISTICS

ENGINE DESIGNATION		ALTERNATIVE	P_c (PSIA)	ϵ	F_{VAC} (KLB)	I_{SPVAC} (SEC)	WEIGHT (LB)	MAXIMUM DIA/LNGTH (IN.)	ENGINE DEV. COST (M $)
J-2		BASIC	780	27.5	230.0	425.0	3450	80/120	22
J-2S		BASIC	1250	40.0	265.0	436.0	3800	80/120	82
J-2	(1)	+ $\Delta\epsilon$	780	34.0	232.2	429.0	3744	91/144	28
J-2S	(A-1)	+ $\Delta\epsilon$	1250	80.0	272.5	448.3	3755	112/176	106
J-2S	(A-2)	+ $\Delta\epsilon$	1250	105.0	275.0	452.8	3855	128/200	108
J-2S	(B-1)	1.22 X P_c BASIC	1520	40.0	320.0	434.5	4120	80/120	107
J-2S	(B-2)	+ $\Delta\epsilon$	1520	80.0	327.5	446.8	4040	112/176	137
J-2S	(B-3)	+ $\Delta\epsilon$	1520	105.0	330.0	451.4	4200	128/200	139
Hi P_C (TYP)		NEW DEVELOPMENT	~3000	90	261	456	2800	75.5/148	444
			~3000	90	320	456	3700	82 /160	511

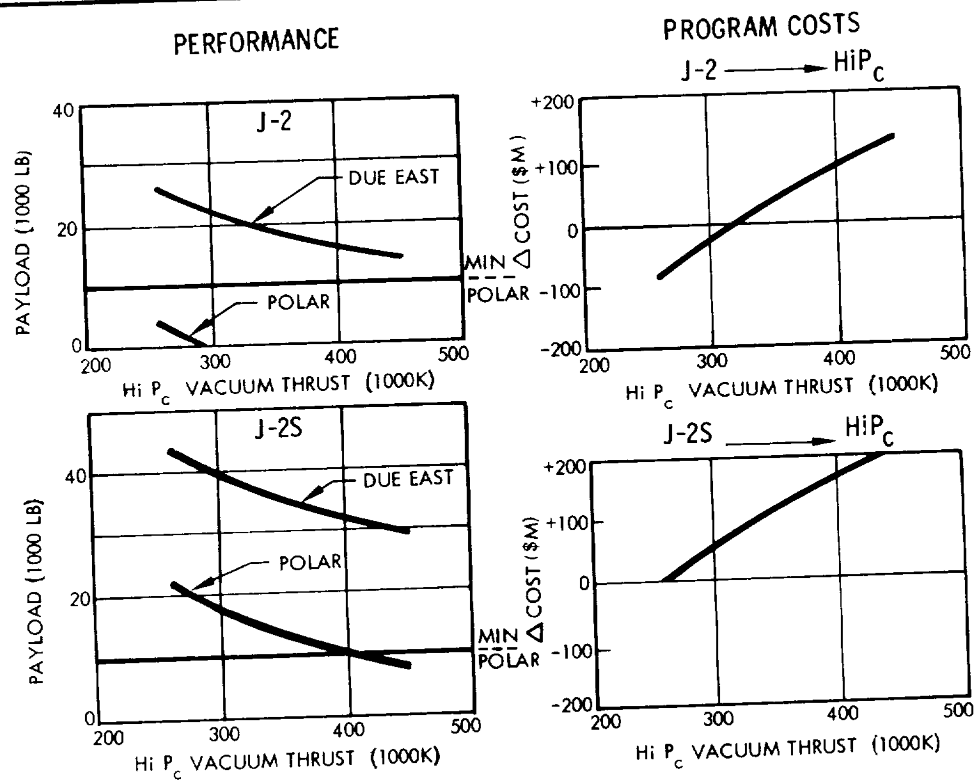
[ ] ENGINES DO NOT SATISFY O4OA INSTALLATION

DO5873

The selection of a HiP_C thrust level, when considerations must be given to Mark I and Mark II capabilities, involves the trade of performance and cost. The formulation logic selected is to size external tanks for the 65K lb payload due east using a HiP_C engine, and then, with the tanks fixed, determine the payload delivery capability of the Mark I system with J-2 and J-2S engines. As HiP_C thrust level is increased, the improved system performance results in reduction in Mark I payload attributed to the larger thrust level differences between the Mark I and Mark II systems.

To accomplish the stated minimum 10K lb polar payload using the J-2 Basic engine for Mark I, the HiP_C thrust level must be below 220K lb. In the case of the J-2S Basic engine, the maximum allowable HiP_C thrust level becomes 400K lb.

The major issue in cost becomes the HiP_C thrust selected because of the relative insensitivity of cost to exterior tank size.


PERFORMANCE
PROGRAM COSTS
J-2 → HiP$_C$
J-2
DUE EAST
POLAR
MIN
POLAR
PAYLOAD (1000 LB)
Δ COST ($M)
Hi P$_c$ VACUUM THRUST (1000K)
J-2S
J-2S → HiP$_C$
40
20
0
200
300
400
500
+200
+100
-100
-200

AVAILABILITY/SCHEDULE RISK EVALUATION

Evaluations of risks associated with availability and schedule point to considerable flexibility for this program.

AVAILABILITY/SCHEDULE RISK EVALUATION

J-2 ⟶ HiP_C

J-2S ⟶ HiP_C

FACTS:

- J-2 AND J-2S HAVE DEMONSTRATED THRUST AND I_{SP}. J-2S NEAR QUALIFICATION.
- HiP_C HAS ONLY BEEN THROUGH COMPONENT DEVELOPMENT.
- J-2 OR J-2S COULD BE AVAILABLE FOR MARK I.
- J-2 OR J-2S WILL HAVE TO BE REDESIGNED AND TESTED FOR HIGHER INLET PRESSURES (OTHER CHARTS).

OPINIONS:

- BECAUSE OF J-2/J-2S AVAILABILITY FOR MARK I, SCHEDULE REVISIONS IN HiP_C ENGINE PROGRAM COULD BE ACCOMMODATED IF DESIRED.
- HiP_C COULD BE AVAILABLE ON SCHEDULE FOR MARK II.

D05882

TECHNOLOGY/APPLICATION RISK EVALUATION

Technology and application risk evaluation for these programs indicate relatively moderate risks.

J-2 → HiP_C

J-2S → HiP_C

FACTS:

- J-2 AND J-2S ENGINES ARE DESIGNED FOR MAXIMUM LO_2 INLET PRESSURE OF 132 PSIA. CURRENT ORBITER DESIGN RESULTS IN INLET PRESSURE OF APPROXIMATELY 200 PSIA.
- J-2 AND J-2S ENGINES ARE SENSITIVE TO INSTABILITY RESULTING FROM PRESSURE FLUCTUATIONS AT START.
- HiP_C ENGINE WILL UTILIZE NEW TECHNOLOGY.
- CHANGE FROM J-2/J-2S TO HiP_C CANNOT BE SIMPLE "PLUG IN." SUBSYSTEM CHANGES IN POGO SUPPRESSION, HYDRAULICS, AVIONICS, GAS SUPPLY, ETC.

OPINIONS

- EARLY AVAILABILITY OF J-2/J-2S FOR ENGINE INSTALLATION AND SUBSYSTEM DEVELOPMENT DECREASES RISK.
- J-2 AND J-2S ENGINES COULD BE SENSITIVE TO STABILITY PROBLEMS WITH LONG OXYGEN FEEDLINES.

COST RISK EVALUATION

The risks related to costs are moderate for this program because the J-2 and J-2S costs are relatively well known. High-pressure engine costs are not proven and are subject to numerous variables. LMSC has the opinion that the incremental component development of the HiP_C ("Breadboard") approach may increase costs as a result of engine-integration problems and overhead costs.

COST RISK EVALUATION

J-2 ⟶ HiP_C

J-2S ⟶ HiP_C

FACTS:

- J-2 COSTS ARE RELATIVELY WELL KNOWN, EXCEPTING OPERATIONAL COSTS.
- J-2S BASIC COSTS ARE RELATIVELY WELL KNOWN, EXCEPTING OPERATIONAL COSTS.
- CURRENT J-2/J-2S COSTS DO NOT INCLUDE MODIFICATION FOR HIGHER INLET PRESSURE REQUIREMENTS (OTHER CHARTS).

OPINIONS:

- COST OF THE HiP_C PROGRAM IS SUBJECT TO NUMEROUS VARIABLES.
- HiP_C INCREMENTAL COMPONENT DEVELOPMENT APPROACH ("BREADBOARD") MAY INCREASE COSTS.
- ADVANCED J-2S OPTIONS WILL HAVE HIGHER DEVELOPMENT COSTS THAN USED IN THIS EVALUATION, BUT EVEN A 50 PERCENT INCREASE IS NOT SIGNIFICANT TO OVERALL PROGRAM COST.

OPERATIONAL RISK EVALUATION

The operational risk evaluation for the program is relatively low, since use of the Hi P_c in the main operational phase of the program provides an engine designed for reuse.

OPERATIONAL RISK EVALUATION

J-2 ⟶ HiP_C

J-2S ⟶ HiP_C

FACTS:

- NEITHER J-2 OR J-2S ARE DESIGNED TO BE REUSABLE ENGINES.
- J-2S HAS MORE DESIRABLE REUSABILITY FEATURES THAN J-2.
- HiP_C ENGINE WILL BE DESIGNED FOR REUSABLE APPLICATION.

OPINION:

- J-2 AND J-2S MINIMUM LIFETIMES ARE KNOWN:

  SCHEDULED MAINTENANCE AT 3000 SEC [7 MISSIONS] OR 20 STARTS

  OVERHAUL AT 6000 SEC [14 MISSIONS] OR 40 STARTS

DO5886

AGENDA

INTRODUCTION

ORBITER & TANKS

- H VS HO
- TANDEM VS PARALLEL
- INTERSTAGE CONCEPTS
- TANK SEPARATION & DISPOSAL
- TANK DESIGN/COST

MARK I/MARK II ENGINE BASELINE

- J-2 OR J-2S → HiP_C
- J-2 OR J-2S FOR MK I/MK II
- HiP_C FOR MK I/MK II
- ENGINE PROGRAM RECOMMENDATION

SUMMARY OF BOOSTER STUDIES

MARK I/MARK II SUBSYSTEM STATUS

- AVIONICS
- TPS
- STRUCTURE (ALUMINUM VS TITANIUM)

The chart opposite summarizes the basic characteristics of the O4OA Orbiter system which employs either the J-2 or J-2S engines or their options in a Mark II as well as Mark I system.

These systems are characterized by low ignition thrust-to-weight ratios (0.6 to 0.3) and external tank propellant requirements in the order of one million pounds. The influence of increased staging velocity from 6000 to 7000 fps is shown as an increase in thrust-to-weight ratio of 0.1 and a reduction in propellant and OLOW in the order of 200,000 lb.

The large values for OLOW and low thrust-to-weight ratios for the J-2 Basic, the J-2S Basic and the J-2S Option 1 are incompatible with the performance capability of the RS-1C booster, and could not be used with that booster as presently defined for Mark II payload requirements. They might be used with another type of booster, however, with a corresponding booster size penalty. The other options (J-2S/A-1, J-2S/A-2, J-2S/B-1, J-2S/B-2, J-2S/B-3) are compatible with the RS-1C booster.

J-2 → J-2
J-2S → J-2S

65K PAYLOAD - DUE EAST

MARK II ENGINE	V_{ST} = 6000 FPS			V_{ST} = 7000 FPS		
	T / W	EXTERNAL TANK PROPELLANT (10^6 LB)	OLOW (10^6 LB)	T / W	EXTERNAL TANK PROPELLANT (10^6 LB)	OLOW (10^6 LB)
J-2 BASIC	0.64	1.15	1.44	0.75	0.94	1.22
J-2S BASIC	0.80	1.04	1.33	0.93	0.86	1.14
J-2 OPT 1	0.66	1.12	1.41	0.77	0.92	1.21
J-2S/A-1	0.89	0.94	1.23			
J-2S/A-2	0.91	0.93	1.21			
J-2S/B-1	1.0	1.0	1.29	1.14	0.84	1.13
J-2S/B-2	1.09	0.92	1.20			
J-2S/B-3	1.12	0.90	1.18			

COST COMPARISONS

The same baseline for cost comparisons was used for all of the programs. This was J-2S Hi P_c as previously defined. The costs shown indicate that the J-2 versions produce the lowest program costs. The J-2S Basic produces the lowest program costs for the J-2S versions. The J-2S B-1 version is very compatible with the O4OA Orbiter base area and has an overall program cost of $42M more than the J-2S Basic.

COST COMPARISONS

J-2 (TWO VERSIONS) $\xrightarrow[\text{ENGINE}]{\text{SAME}}$ J-2 (TWO VERSIONS)

J-2S (SIX VERSIONS) $\xrightarrow[\text{ENGINE}]{\text{SAME}}$ J-2S (SIX VERSIONS)

ENGINE	TANKS N.R.	TANKS REC.	ENGINE N.R.	ENGINE REC.	TOTAL N.R.	TOTAL REC.	TOTAL	DELTA FROM BASELINE ($M)
BASELINE *					693	776	1469	
J-2 BASIC	209	607	22	257	231	864	1095	-374
J-2S BASIC	204	582	82	350	286	940	1226	-243
J-2 OPTION 1	208	603	28	260	236	863	1099	-370
J-2S A-1	198	549	106	385	304	934	1238	-231
J-2S A-2	198	542	108	385	306	927	1233	-236
J-2S B-1	202	566	107	393	309	959	1268	-201
J-2S B-2	197	538	137	400	334	938	1272	-197
J-2S B-3	191	533	139	400	330	933	1263	-206

* BASELINE J2S → HiP$_C$

DO5937

AVAILABILITY/SCHEDULE RISK EVALUATION

The availability and schedule risk of the J-2 or J-2S engines are the lowest of any of the proposed programs. These engines can be available for Mark I and continue through Mark II.

AVAILABILITY/SCHEDULE RISK EVALUATION

$$\text{J-2} \xrightarrow[\text{ENGINE}]{\text{SAME}} \text{J-2}$$

$$\text{J-2S} \xrightarrow[\text{ENGINE}]{\text{SAME}} \text{J-2S}$$

FACTS:

- J-2 AND J-2S HAVE DEMONSTRATED THRUST AND I_{SP}. J-2S NEAR QUALIFICATION.
- J-2 OR J-2S CAN BE AVAILABLE FOR MARK I
- J-2 OR J-2S WILL HAVE TO BE REDESIGNED AND TESTED FOR HIGHER INLET PRESSURES (OTHER CHARTS).

OPINION:

- J-2 OPTIONS REQUIRING EXPANSION RATIO OR CHAMBER PRESSURE CHANGES COULD BE AVAILABLE FOR MARK I.

TECHNOLOGY/APPLICATION RISK EVALUATION

The technology and application risks associated with this program are relatively low.

$$\text{J-2} \xrightarrow[\text{ENGINE}]{\text{SAME}} \text{J-2}$$

$$\text{J-2S} \xrightarrow[\text{ENGINE}]{\text{SAME}} \text{J-2S}$$

FACTS:

- J-2 AND J-2S ENGINES ARE DESIGNED FOR MAXIMUM LO_2 INLET PRESSURES OF 132 PSIA. CURRENT ORBITER DESIGN RESULTS IN INLET PRESSURE OF APPROXIMATELY 200 PSIA.
- J-2 AND J-2S ENGINES ARE SENSITIVE TO INSTABILITY RESULTING FROM PRESSURE FLUCTUATIONS AT START.

OPINION:

- J-2 AND J-2S ENGINES COULD BE SENSITIVE TO STABILITY PROBLEMS WITH LONG OXYGEN FEEDLINES.

COST RISK EVALUATION

The J-2 and J-2S costs are based on recent experience and are considered to present the lowest cost risk.

$$\text{J-2} \xrightarrow[\text{ENGINE}]{\text{SAME}} \text{J-2}$$

$$\text{J-2S} \xrightarrow[\text{ENGINE}]{\text{SAME}} \text{J-2S}$$

FACTS:

- J-2 COSTS ARE RELATIVELY WELL KNOWN, EXCEPTING OPERATIONAL COSTS.
- J-2S BASIC COSTS ARE RELATIVELY WELL KNOWN, EXCEPTING OPERATIONAL COSTS.
- CURRENT J-2/J-2S COSTS DO NOT INCLUDE MODIFICATIONS FOR HIGHER INLET PRESSURE REQUIREMENTS (OTHER CHARTS).

OPINION:

- ADVANCED J-2S OPTIONS WILL HAVE HIGHER DEVELOPMENT COSTS THAN USED IN THIS EVALUATION, BUT EVEN A 50 PERCENT INCREASE IS NOT SIGNIFICANT TO OVERALL PROGRAM COSTS.

DO5881

OPERATIONAL RISK EVALUATION

Since the engines are not designed for reusable applications, the operational risks are considered to be higher than for programs employing HiP_C engines in Mark II.

J-2 $\xrightarrow[\text{ENGINE}]{\text{SAME}}$ J-2

J-2S $\xrightarrow[\text{ENGINE}]{\text{SAME}}$ J-2S

FACTS:

- NEITHER J-2 OR J-2S ARE DESIGNED TO BE REUSABLE ENGINES.
- J-2S HAS MORE DESIRABLE REUSABILITY FEATURES THAN J-2.
- SUBSYSTEM CHANGES FROM MARK I TO MARK II WILL NOT BE NECESSARY.

OPINION:

- J-2 AND J-2S MINIMUM LIFETIMES ARE KNOWN:

  SCHEDULED MAINTENANCE AT 3000 SEC [7 MISSIONS]

  OVERHAUL AT 6000 SEC [14 MISSIONS] OR 40 STARTS

DO5878

AGENDA

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- J-2 OR J-2S $\longrightarrow$ HiP_C
- J-2 OR J-2S FOR MK I/MK II
- $\boxed{HiP_C \text{ FOR MK I/MK II}}$
- ENGINE PROGRAM RECOMMENDATION

SUMMARY OF BOOSTER STUDIES

MARK I/MARK II SUBSYSTEM STATUS

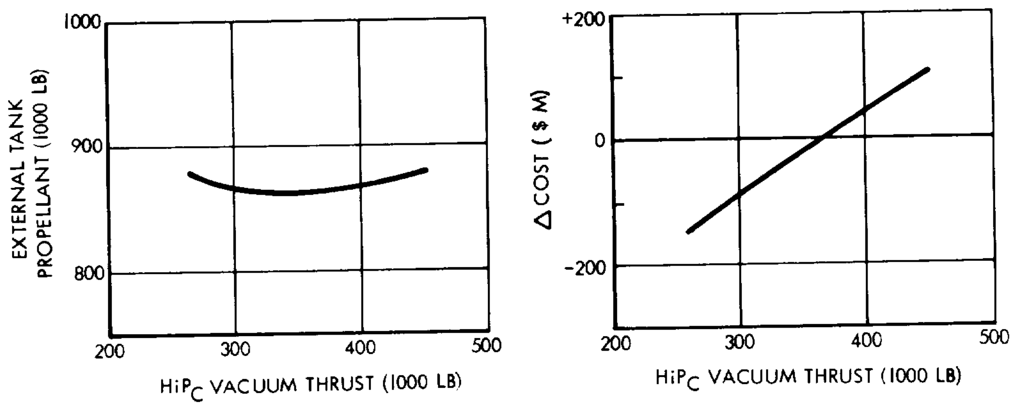
- AVIONICS
- TPS
- STRUCTURE (ALUMINUM VS TITANIUM)

D05786(9)

Summarized in the chart opposite are the pertinent design and cost parameters to the selection of a HiP_C engine for use in both Mark I/Mark II designs. Over the range of thrust levels explored, orbiter weight varied from approximately 120,000 lb at the low thrust level to 140,000 lb at the highest point shown. This weight increase, however, is negated by reduction in velocity loss as orbiter thrust is increased, resulting in external tank propellant requirements that show low sensitivity to thrust level selection. Due to the small differences in exterior tank size as well as their low cost, the primary cost factor, as shown, becomes dependent on the thrust level selected.

PERFORMANCE AND COST
$HiP_C \longrightarrow HiP_C$

65K PAYLOAD - DUE EAST



AVAILABILITY/SCHEDULE RISK EVALUATION

This program results in the highest risk relative to availability and schedule. A new, advanced technology engine would have to be available for the Mark I vehicle.

AVAILABILITY/SCHEDULE RISK EVALUATION

$HiP_C \xrightarrow[\text{ENGINE}]{\text{SAME}} HiP_C$

FACTS:

- HiP_C HAS ONLY BEEN THROUGH COMPONENT DEVELOPMENT.
- HiP_C DEVELOPMENT PROGRAMS ARE BASED UPON EXPERIENCE ONLY IN LOW-PRESSURE ENGINE PROGRAMS.

OPINION:

- HiP_C PROGRAM SCHEDULE PROVIDES LITTLE MARGIN FOR SAFETY IF ENGINE FOR MARK I IS AVAILABLE.

DO5877

TECHNOLOGY/APPLICATION RISK EVALUATION

The technology and application risk for this program will be relatively high as compared to other programs. The new technology engine must be applied to Mark I.

TECHNOLOGY/APPLICATION RISK EVALUATION

$$\text{Hi } P_C \xrightarrow[\text{ENGINE}]{\text{SAME}} \text{Hi } P_C$$

FACTS:

- Hi P_C ENGINE WILL UTILIZE NEW TECHNOLOGY.
- SUBSYSTEM CHANGES FROM MARK I TO MARK II WILL NOT BE NECESSARY.

OPINION:

- ENGINES WILL NOT BE AVAILABLE EARLY FOR ENGINE INSTALLATIONS AND SUBSYSTEM DEVELOPMENT, AND THIS WILL INCREASE RISKS.

DO5883

COST RISK EVALUATION

The basis for costs of the High Pressure program are not well established as compared to the J-2 or J-2S program. This leads to potentially higher risks relative to obtaining a Mark I engine.

COST RISK EVALUATION

$$HiP_C \xrightarrow[\text{ENGINE}]{\text{SAME}} HiP_C$$

FACT:

- HiP_C COSTS ARE EXTRAPOLATED FROM LOW-PRESSURE PROGRAMS.

OPINIONS:

- COST OF THE HiP_C PROGRAM IS SUBJECT TO NUMEROUS VARIABLES.

DO5884

OPERATIONAL RISK EVALUATION

The operational risks associated with employing the HiP_C engine in Mark I and Mark II will be the lowest relative to programs employing J-2 and J-2S. The engine will be designed for the shuttle application.

OPERATIONAL RISK EVALUATION

$HiP_C \longrightarrow HiP_C$

FACT:

- HiP_C ENGINE WILL BE DESIGNED FOR REUSABLE APPLICATION.

OPINION:

- OPERATIONAL COSTS SHOULD BE LOWER THAN PROGRAMS USING J-2 OR J-2S

DO5879

AGENDA

INTRODUCTION

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- TANK DESIGN/COST

MARK I/MARK II ENGINE BASELINE

- J-2 OR J-2S ⟶ HiP_C
- J-2 OR J-2S FOR MK I/MK II
- HiP_C FOR MK I/MK II
- ENGINE PROGRAM RECOMMENDATION

SUMMARY OF BOOSTER STUDIES

MARK I/MARK II SUBSYSTEM STATUS

- AVIONICS
- TPS
- STRUCTURE (ALUMINUM VS TITANIUM)

O4OA SENSITIVITIES

The data presented in the opposite chart is for O4OA system sensitivities to orbiter weight change. The two terms dominating sensitivity are thrust and specific impulse. These data as derived represent a 10,000-lb orbiter-weight increase, and its effect on OLOW and GLOW. To some extent, the orbiter is treated as a frozen design, and weight cascading effects are minimized to reflect changes in thrust structure, landing gear, wing structure, and TPS only. Development of the gross liftoff-to-orbiter weight sensitivity was attained by applying a multiplier of six, derived during Phase B heat-sink booster studies.

040A SENSITIVITIES

65K LB PAYLOAD – DUE EAST

V_{STAGE} = 6000 FPS-REL

COMMON MARK I/MARK II PROPULSION

ENGINE	NOMINAL T/W	∂OLOW/∂ORBITER WT	∂GLOW/∂ORBITER WT
J-2 BASIC	0.64	25.5	153
J-2S BASIC	0.80	12.0	72
HiP_C (T_V = 260K)	0.90	9.2	55
HiP_C (T_V = 360K)	1.26	7.7	46

D05772
llm

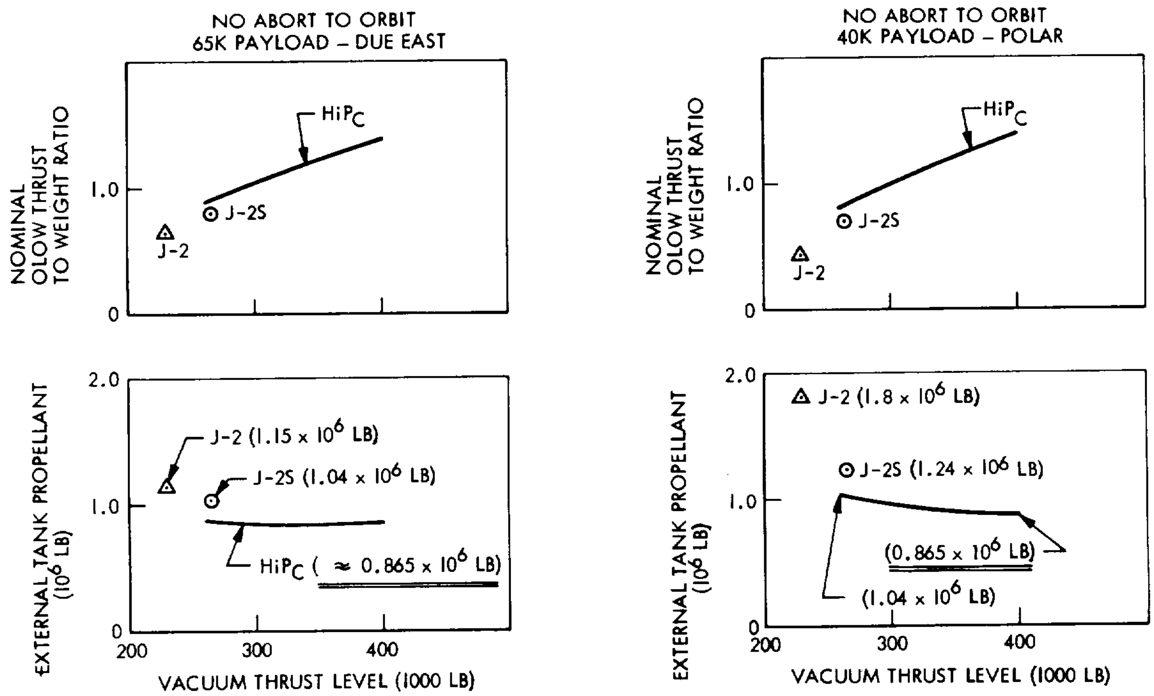
INFLUENCE OF ABORT TO ORBIT

Although presently abort to orbit is not a requirement, it is felt that its effect on system design should be reviewed. The basic difference between designing for, or not designing for, abort capability is that of critical mission definition. The due east mission, due to the combination of payload and 1000 fps on-orbit velocity requirements, is the critical nominal mission. In the case of abort to orbit, the polar mission is critical, since higher velocity losses occur when an orbiter engine is lost at staging and only 650 ft/sec of orbit-maneuvering capability is available.

The effects of these considerations are summarized in terms of tank size and orbiter (all engines operating) thrust-to-weight ratio. Significant to the system influence of accommodating for abort to orbit are the thrust-to-weight ratio and specific impulse characteristics developed around the propulsion system under investigation. The J-2 system, which has the lowest thrust-to-weight ratio and specific impulse, shows the greatest sensitivity to designing for abort to orbit. As indicated by the bracketed data, exterior tank propellant increases from 1.15×10^6 lb for nominal mission design to 1.8×10^6 lb for abort to orbit design. This effect is reduced as engine performance improves to the point where the 400K HiP_C engine shows the same propellant requirement for the nominal as well as the aborted mission.

INFLUENCE OF ABORT TO ORBIT

COMMON MARK I/MARK II ENGINES



SUMMARY REGARDING ENGINE PROGRAM SELECTION

This table summarizes the principal differences among the engine selection options discussed in the previous chart. A significant factor not shown in the table is the compatability with the boost capability of the proposed RS-1C booster. The RS-1C has excess capability for the configurations employing the HiP_C engine but underperforms for the J-2 → J-2 and the J-2S → J-2S systems. Since the J2 → HiP_C option does not meet the minimum Mark I payload requirement, only the J-2S → HiP_C and the HiP_C → HiP_C options are available. Of these, Lockheed recommends the J-2S → HiP_C because of the lower risk and because of the better growth capability afforded by the HiP_C engine in Mark II.

SUMMARY REGARDING ENGINE PROGRAM SELECTION

ABORT TO ORBIT NOT CONSIDERED

OPTION	RELATIVE COST ($ MILLIONS)	ENGINE PROGRAM RISK	SYSTEM SENSITIVITY ∂ GLOW/ ∂ ORBITER WEIGHT	MARK I PAYLOAD	EFFECT ON PEAK ANNUAL FUNDING ($ MILLIONS)
J-2 $\longrightarrow$ HiP_c	-100	MODERATE	-	5000 LB POLAR 27K DUE EAST (250K Hi Pc)	-18
J-2S $\longrightarrow$ HiP_c	BASELINE	MODERATE	-	23,000 LB POLAR 45K DUE EAST (250K Hi Pc)	0
J-2 $\xrightarrow[\text{ENGINE}]{\text{SAME}}$ J-2	-374	LOW TO MODERATE	153	40K POLAR 65K DUE EAST	-15
J-2S $\xrightarrow[\text{ENGINE}]{\text{SAME}}$ J-2S	-243	LOWEST	72	40K POLAR 65K DUE EAST	-15
HiP_c $\xrightarrow[\text{ENGINE}]{\text{SAME}}$ HiP_c	-150	HIGHEST	55 (260K) 46 (360K)	40K POLAR 65K DUE EAST	+100

AGENDA

INTRODUCTION

ORBITER & TANKS

- H VS HO
- TANDEM VS PARALLEL
- INTERSTAGE CONCEPTS
- TANK SEPARATION & DISPOSAL
- TANK DESIGN/COST

MARK I/MARK II ENGINE BASELINE

- J-2 OR J-2S ⟶ HiP_C
- J-2 OR J-2S FOR MK I/MK II
- HiP_C FOR MK I/MK II
- ENGINE PROGRAM RECOMMENDATION

SUMMARY OF BOOSTER STUDIES

MARK I/MARK II SUBSYSTEM STATUS

- AVIONICS
- TPS
- STRUCTURE (ALUMINUM VS TITANIUM)

After completion of the initial sizing studies, a baseline SRM-stage configuration, consisting of a single-stage SRM-booster with four 156-in. dia solid rocket motors (SRM), was selected. The subsequent activities were as summarized on the chart opposite, and consisted of three activity groupings: (1) study of performance characteristics and cost estimates, (2) preliminary design and cost estimates for the stage subsystems, and (3) analysis of the system potentials based on projections from the present studies.

Since the level of effort on SRM-boosters was reduced after the midterm briefing (by NASA direction), the study activity was based on extension of the original work completed. Therefore, the data presented herein do not conform with the latest groundrules for the remainder of the study. Except for some work presented in regard to future potentials for an SRM-booster which is employed as a final operational booster, the work represented presumes utilization of the SRM as an interim booster for 12 operational launches.

SRM-STAGE ACTIVITIES

SOLID ROCKET MOTORS (SRMs)

- PERFORMANCE DATA COMPARISONS
- COST DATA COMPARISONS
- SUBCONTRACTOR STUDY - DATA CRITIQUES
- DOCUMENTATION OF SUBCONTRACTOR STUDIES

SRM STAGES

- STAGE WEIGHTS AND COSTS
- SRM STAGE MANRATING REQUIREMENTS
- RECOVERY AND REFURBISHMENT POTENTIALS
- PROGRAM COSTS

SRM BOOSTER SYSTEM STUDIES

- SRM BOOSTER COST-EFFECTIVENESS POTENTIALS
- SYSTEM UNCERTAINTY AREAS
- FUTURE STUDY NEEDS FOR SRM BOOSTERS

SRM SUBCONTRACTOR STUDIES

The bulk of the large solid motor technology has been collectively developed by four major propulsion companies. A working relationship was established and maintained throughout this study to ensure that the SRM data being used were current and valid, and represented the judgment of the most knowledgeable sources. The four companies were funded the latter part of August to generate and provide to this study specific design and cost parametric data that could be used to establish realistic SRM motor designs and the respective development and recurring costs.

Preliminary submittals of data have been received, analyzed, and integrated into the LMSC study effort. Final reports from the SRM contractors are scheduled for mid-November. Engineering Memorandums have been prepared which present and evaluate the contractor data submittals.

A primary goal of the four subcontract studies was to reduce uncertainty ranges on both performance and cost parameters and, thereby, increase the confidence in the maturity of SRM state-of-the-art.

SRM SUBCONTRACTOR STUDIES

PURPOSE: TO OBTAIN COMPARATIVE PARAMETRIC DESIGN, PERFORMANCE, AND COST DATA FOR SRMs FOR POTENTIAL SPACE SHUTTLE APPLICATIONS

MOTOR SIZES	COMPANY	GO-AHEAD DATE	INITIAL INPUT	INTERIM REPORTS	FINAL REPORTS
156/260 IN.	AEROJET GENERAL (AGC)	8-24	9-14	10-22	11-15
120/156 IN.	LOCKHEED PROPULSION (LPC)				
120/156/260 IN.	THIOKOL CHEMICAL (TCC)				
120/156 IN.	UNITED TECHNOLOGY (UTC)	8-31	9-21	——	10-22

DESIGN DATA

- MASS FRACTIONS
- SPECIFIC IMPULSE
- MOTOR LENGTHS
- 3 MEOPs
- TWO BURN TIMES
- TYPICAL MOTORS
- SPECIAL DESIGN TRADEOFFS

COST DATA

- NONRECURRING: 8 & 12 MOTOR FIRINGS
- RECURRING: 20M & 40M LB/YR PROPELLANT PRODUCTION
- SPECIAL FACTOR INFLUENCES ON COST

DO5595 (I)

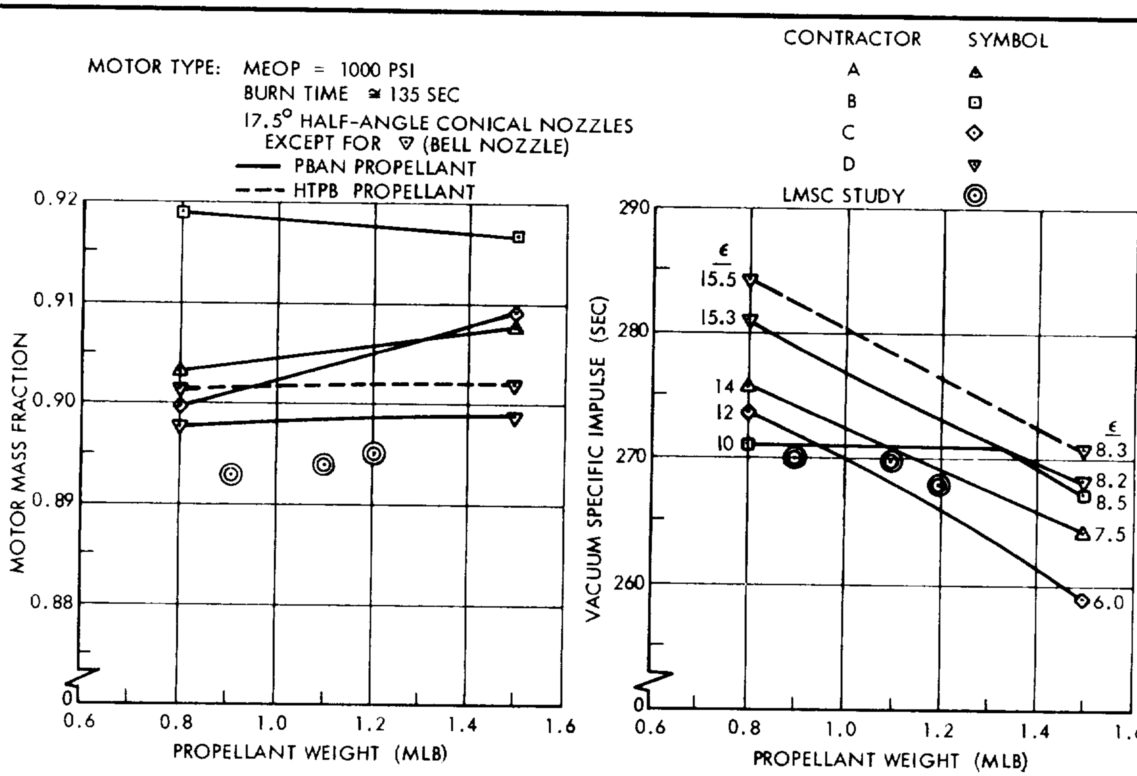
Motor performance parametrics are shown, as applicable for LMSC alternate concepts study baseline single-stage boosters. The motor designs utilize D6AC steel, as used in the Titan IIIC/D 120-in. dia motors, in the solid rocket motor case. All other elements of the motor design represent use of established state-of-the-art materials and fabrication practices. The dashline data represent a new propellant currently being utilized in advanced missile development programs, namely, the HTPB formulation (Hydroxyl terminated polybutadiene).

The mass fractions, as presented by three of the motor subcontractors, are very close (plus or minus 0.004); one contractor chose more optimistic data by about plus 0.015. In regard to mass fraction, LMSC baseline performance and sizing appears conservative by about minus 0.01.

The vacuum I_{sp} values are within about plus or minus 3 sec, again except for one subcontractor who shows about 4 sec higher values. The LMSC study values of vacuum I_{sp} are conservative by about plus or minus 3 sec.

The HTPB propellants show improvements both in mass fraction (due to the higher propellant densities achievable) and in the I_{sp} values attainable. All subcontractors agree that cost is affected negligibly (or reduced) when this improved propellant is employed. The reduced SRM stage sizes made possible by both the conservatisms discussed above and by use of HTPB have not as yet been introduced in the SRM-booster systems cost-effectiveness analyses.

156-IN. SRM PERFORMANCE COMPARISON



D04875(1)

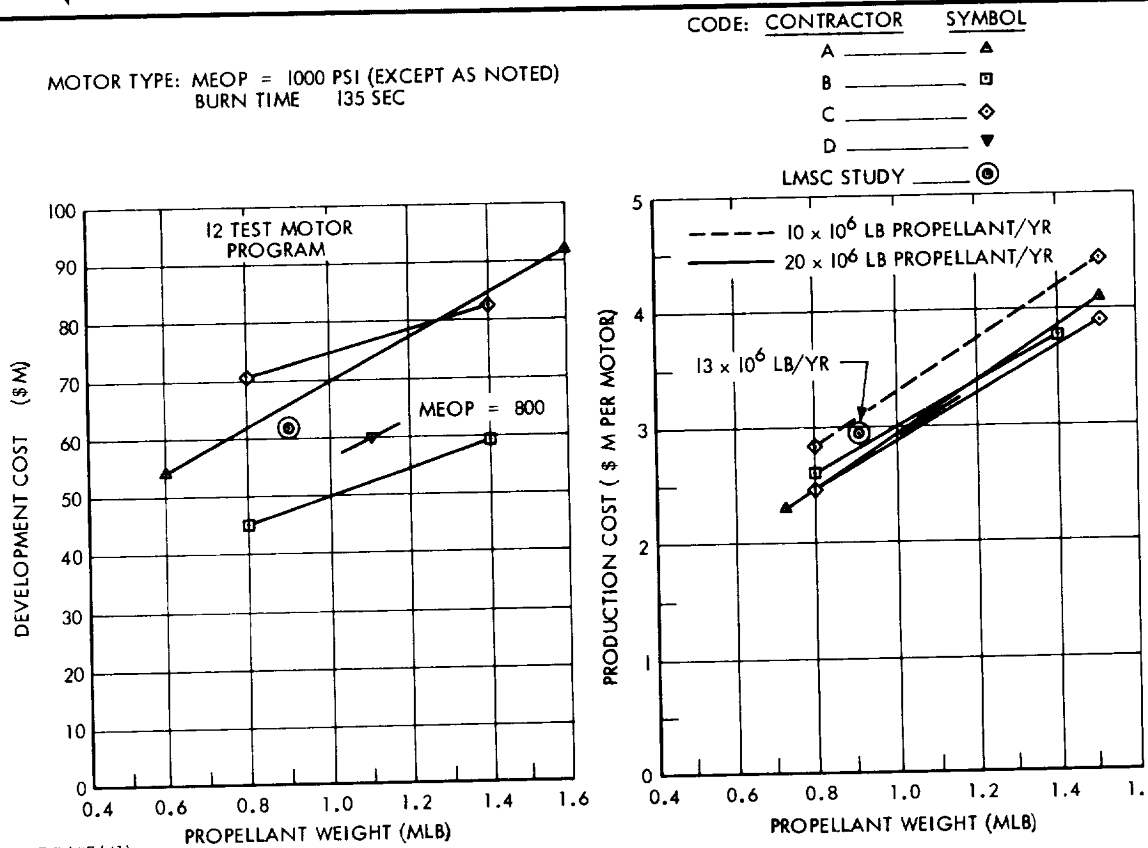
156-IN. SRM COST COMPARISON

The four SRM subcontractors prepared parametric cost data, which reflect the effects of the major motor design parameters. These data are representative for the design regime applicable to a single-stage SRM-booster system. The chart presents both development and production costs for motors used in the baseline study.

Development costs are shown for a 12-motor test-firing program, which is probably more than is required (ten test firings were recommended by NASA/MSFC). On development costs, three of the subcontractors straddle the 62 million dollar value used in the LMSC system cost analysis. Again, one subcontractor showed optimistic values about 25 percent lower. This spread is due in part to differences between subcontractor assumptions in including costs for facilities, tooling, transportation equipment, and GSE in either nonrecurring or recurring cost breakouts. More uniform breakout is expected in the subcontractors' final reports. The LMSC study development cost value is quite conservative if fewer test motors are required.

Production costs per motor are well within plus or minus 5 percent, in the cost data provided by all four subcontractors. The LMSC study value is shown at a production rate (13×10^6 lb per year) comparable to four launches per year. Prior to receipt of the subcontractor data, LMSC used for the study baseline a value of 3.4 million dollars in lieu of the present value of 2.9 million dollars. Nonetheless, the present value is considered conservative by about 10 percent (high).

156-IN. SRM COST COMPARISON



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The SRM-subcontractor studies have produced a remarkable consistency of both performance and cost characteristics in all three sizes of large solid rockets (120 in., 156 in., and 260 in. diameters). Three of the subcontractors independently provided essentially identical characteristics, allowing for a rather narrow band of variation, in the area of noise-level differences. One of the subcontractors provided appreciably more optimistic characteristics, both in mass-fraction (performance) and in development costs. This contractor, however, has extensive experience in developing and producing a wide variety of experimental and operational solid rockets, and has a proven history of "ability to deliver" both in developmental and production programs.

Using data represented by the average of the three similar sets of characteristics (excluding the apparently optimistic set), the SRM-performance characteristics used by Lockheed in the Space Shuttle System studies must be upgraded. With the data now available, the selection of a most cost-effective MEOP still remains to be optimized. When these changes are incorporated, a cost saving of approximately 10 percent is expected in the SRM-booster stage in the program costs shown in separate charts. Also, concurrence exists among the SRM contractors that the HTPB (Hydroxyl Terminated Polybutediene) propellants are now "state-of-the-art" and planned for on-going missile programs. Because of their higher density, improved specific impulse, and small change in costs, additional reductions in SRM-stage cost of approximately 10 percent can be expected.

The study results indicate that the SRM technology required for the space shuttle is proven state-of-the-art, with very low risk levels in achieving both performance and cost goals. Lockheed is convinced that this technology (for the 120-in. and 156-in. motor sizes) can be realistically procured by standard competitive fixed-price contract procedures.

RECAP OF SRM STUDIES

DATA CONSISTENCY: BETTER THAN EXPECTED

- MASS FRACTIONS – 3 VERY CLOSE, 1 AT 0.015 HIGHER
- SPECIFIC IMPULSE (VACUUM) – ALL WITHIN ±3 SEC
- DEVELOPMENT COSTS – 3 VERY CLOSE, 1 AT 30 PERCENT LOWER
- PRODUCTION COSTS – ALL WITHIN ±5 PERCENT

INTERFACE PROBLEM AREAS: REQUIRES CLARIFICATION IN SRM SPECS

- DEFINITION OF DELIVERED ISP
- WEIGHT AND COST FOR THRUST TERMINATION AND TVC ACTUATION
- SAFETY FACTOR CRITERIA
- TOOLING AND FACILITY COSTS BETWEEN DEVELOPMENT AND PRODUCTION

NUMERICAL VALUE RESULTS

LMSC INITIAL VALUES		RECOMMENDED SRM VALUES
0.893	SRM MASS FRACTION	0.900
270	ISP_{VAC} (SEC)	273*
\$62M	DEVELOPMENT COST	\$62M
\$3.4M	PRODUCTION COST/SRM	\$2.9M

*MAY GO UP TO 277 IF HTPB IS USED

D05679

The chart opposite summarizes the total SRM single-stage booster costs, for the earlier groundrules of a 12-vehicle interim-booster operational program. The program costs reflect the previously discussed SRM-stage development and recurring production costs, plus the stage qualification ground and flight test program costs. Launch facility modification and construction costs, normally covered elsewhere in totaling Space Shuttle program costs, are not included. Second-level breakouts of the ground and flight test programs were presented at the mid-term briefing.

For the groundrules specified by NASA, and the SRM-stage configuration and development program philosophy represented in this study, this total-program estimate is considered accurate within ±10 percent.

SRM – BOOSTER PROGRAM COSTS

NONRECURRING ITEMS	COST ($M)
SRM DDT&E*	$ 62
STAGE SUBSYSTEMS DDT&E	22
STAGE INTEGRATION AND SYSTEM SUPPORT	13
STAGE GROUND-TEST PROGRAM*	15
STAGE FLIGHT-TEST PROGRAM	33
NONRECURRING COST TOTAL	$ 145
RECURRING ITEMS	
SRMs (11.6M x 12)	$ 139
STAGE SUBSYSTEMS (7.4M x 12)	89
RECURRING COST TOTAL	$ 228
TOTAL PROGRAM COST**	$ 373

NOTES: * INCLUDES FACILITIES, TOOLING, AND FIXTURES
** BASED ON 12 INTERIM SYSTEM LAUNCHES AND ONE UNMANNED BOOSTER FLIGHT TEST

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rwg

The chart opposite presents manrating design criteria for an SRM-stage. Primarily, previous efforts to establish requirements for manrating solid-propellant motors were associated with the Air Force MOL program. Manrating the SRM-stage involves essentially three aspects: (1) the determination and use of adequate factors of safety for the static components of the bare motor; (2) the use of redundant components for dynamic systems, such as the TVC power train, motor-igniter initiators, thrust-termination initiators and cutting ordinance, and separation ordinance and motors; and (3) an appropriate program of qualification testing of the SRMs and the assembled SRM-stage.

Special incorporated design features would enable rigorous NDT inspection of static components, and the sensing of impending or commencing motor failure. Thrust-termination is required to shut off SRMs in the event that abort procedure is dictated by events occurring anywhere in the Space Shuttle system. All test motors are instrumented to monitor stress and thermal history of critical components to ensure that the design margins-of-safety are being achieved. Proposed safety factors, which match currently used manrating requirements for the liquid-propellant stages, are in excess of the factors used in prior SRM applications that have already demonstrated an extremely high reliability status.

SRM-STAGE MANRATING REQUIREMENTS

SPECIAL DESIGN FEATURES

- THRUST TERMINATION ON SRMs
- LAUNCH-HOLD TO T-1 SEC
- SRM PRESSURE ANOMALY SENSORS
- RIGOROUS NDT ACCEPTANCE PROCEDURES

REDUNDANT SUBSYSTEMS

- TVC ACTUATION
- SRM IGNITION INITIATORS AND COMMAND
- THRUST TERMINATION INITIATORS
- STAGING INITIATORS AND COMMAND

SRM DEVEL/QUAL TESTS

- 3 DEVEL. (FOR NEW 120 IN. SRM)
- 5 PFRT (FOR NEW 120 IN. SRM)
- 156 IN. MAY REQUIRE 9 TO 12 FIRINGS
- 260 IN. MAY REQUIRE 10 TO 15 FIRINGS

SAFETY FACTORS

- 1.15 PROOF TEST (ON MEOP)
- 1.4 ULT. (ON MEOP)
- 2.0 (ON THICKNESS) NOZZLE ABLATOR
- 2.0 (ON THICKNESS) CASE INSULATION
- 1.4 ULT. ON INTERSTAGE STRUCTURES

STAGE QUALIFICATION

- FULL-SCALE STATIC/DYNAMIC TEST
- ASSEMBLY/CHECKOUT OPS DEMONSTRATION
- ONE UNMANNED FLIGHT TEST (WITHOUT ORBITER)*
- SUBSYSTEM QUAL TESTS

*ASSUMED REQUIRED FOR ANY BOOSTER PRIOR TO FLIGHT WITH EXPENSIVE ORBITER

Consideration of use of a solid propellant rocket motor stage for a large number of launches raises the possibility of significant cost savings by recovery and refurbishment of major portions of such a stage. Refurbishment and multiple reuse of rocket motor components have been standard practice in solid rocket motor test programs and have been successfully accomplished in the 120, 156, and 260-in. diameter motor development programs.

All structural components are thermally protected so that no degradation occurs during the motor operation and, therefore are "good-as-new" for reuse. The effect of short-time immersion in seawater should be comparable to the long term saltwater environment to which Fleet Ballistic Missile motors, and high-strength steel hulls of deep submergence vehicles, have been exposed without deleterious effects. The remaining question to refurbishing and reuse is the possibility of structural damage upon impact with the water.

A study by the National Engineering Science Corporation on booster recovery at sea* concludes that recovery can be successfully accomplished without structural damage if minor design requirements are incorporated in the motor and the motor entry into the water is maintained near vertical by the descent recovery equipment. Additional study is required to evaluate the cost of descent recovery systems, retrieval and transport, and refurbishing.

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*"Recovery of Boosters at Sea," National Engineering Science Corporation, Pasadena, California, Report No. S-260, April 1967, under Contract NAS 7-394.

RECOVERY AND REFURBISHMENT POTENTIALS

REUSABLE COMPONENTS

SRM
- MOTOR CASE
- NOZZLE FLEX JOINT
- THRUST TERMINATION SYSTEM
- NOZZLE STEEL SHELL

STAGE
- 50 PERCENT (APPROXIMATELY) OF STRUCTURE
- MOST OF ELECTRICAL SYSTEM
- TVC ACTUATION SYSTEM
- 50 PERCENT (APPROXIMATELY) OF RECOVERY SYSTEM

REPLACED/REFURBISHED ELEMENTS

SRM
- PROPELLANT AND IGNITER
- CASE INSULATOR
- NOZZLE ABLATOR

STAGE
- INTERSTAGE (TO BOOSTER)
- STAGING JOINTS AND ROCKETS
- PART OF RECOVERY SYSTEM

ESTIMATED COST SAVINGS

FROM 20 TO 40 PERCENT REDUCTION OF RECURRING SRM BOOSTER COSTS

D05661
rwg

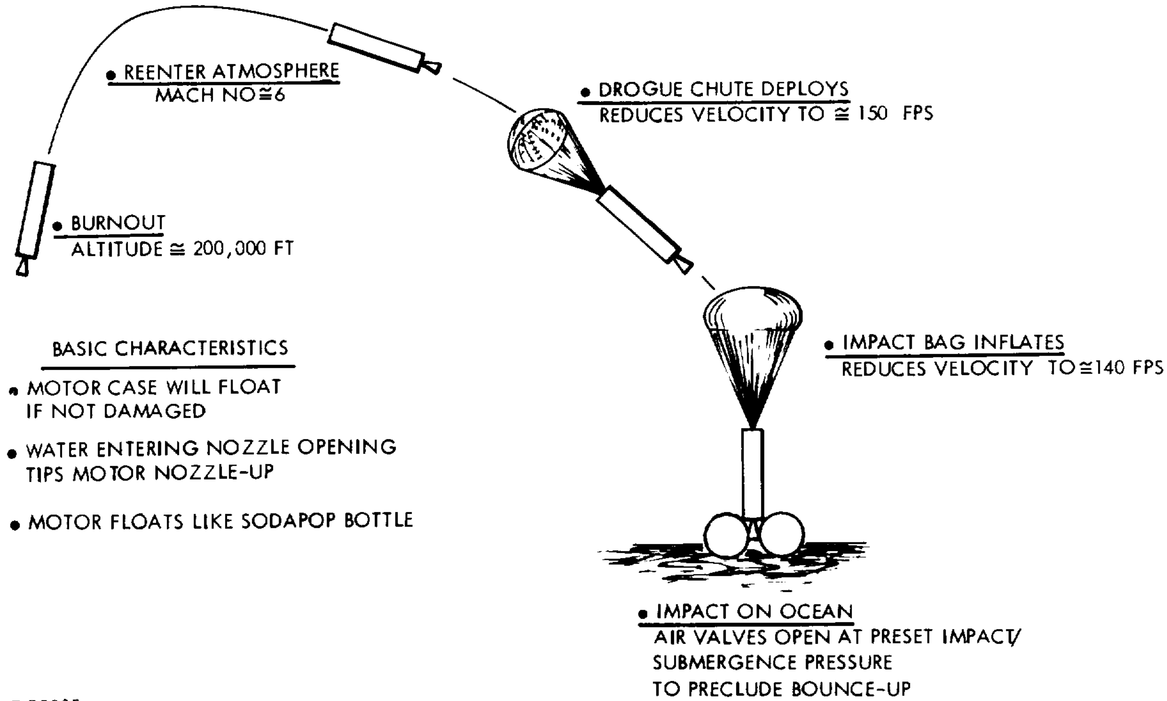
SRM BOOSTER RECOVERY CONCEPT

The chart opposite illustrates one concept for recovery of major segments of an SRM-booster. In the case of the Configuration 5B SRM-stage consisting of three SRMs, the illustration represents the sequence for each of the three primary motors, to which are attached reusable elements of interstage structure and subsystems. Such a recoverable stage would incorporate separation joint designs which would leave most of the interstage elements attached to the SRMs, with a minimum of quick-replacement components at the separation joints.

Electrical wiring and control system components would be potted to assure survival in the recovery-phase environment. The cold-gas storage TVC actuation system is of a basically rugged design, and can readily be configured to require simply a gas recharge as part of the next preflight assembly and checkout operation. SRM case insulation and nozzle ablator will require refurbishment, prior to reloading the motor with the solid propellant grain. The flex-joint assembly, between the nozzle and the motor case, is expected only to require a replacement of the joint thermal-seal membrane.

Recovery system components will require some refurbishment, some replacement, and re-packaging. The one component which requires peculiarly tailored design and development testing is the impact bag and its pressure-relief valving.

SRM BOOSTER RECOVERY CONCEPT



BASIC CHARACTERISTICS

- MOTOR CASE WILL FLOAT IF NOT DAMAGED
- WATER ENTERING NOZZLE OPENING TIPS MOTOR NOZZLE-UP
- MOTOR FLOATS LIKE SODAPOP BOTTLE

DO5905

AGENDA

INTRODUCTION

ORBITER & TANKS

- H VS HO
- TANDEM VS PARALLEL
- INTERSTAGE CONCEPTS
- TANK SEPARATION & DISPOSAL
- TANK DESIGN/COST

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- J-2 OR J-2S $\longrightarrow$ HiP_C
- J-2 OR J-2S FOR MK I/MK II
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- ENGINE PROGRAM RECOMMENDATION

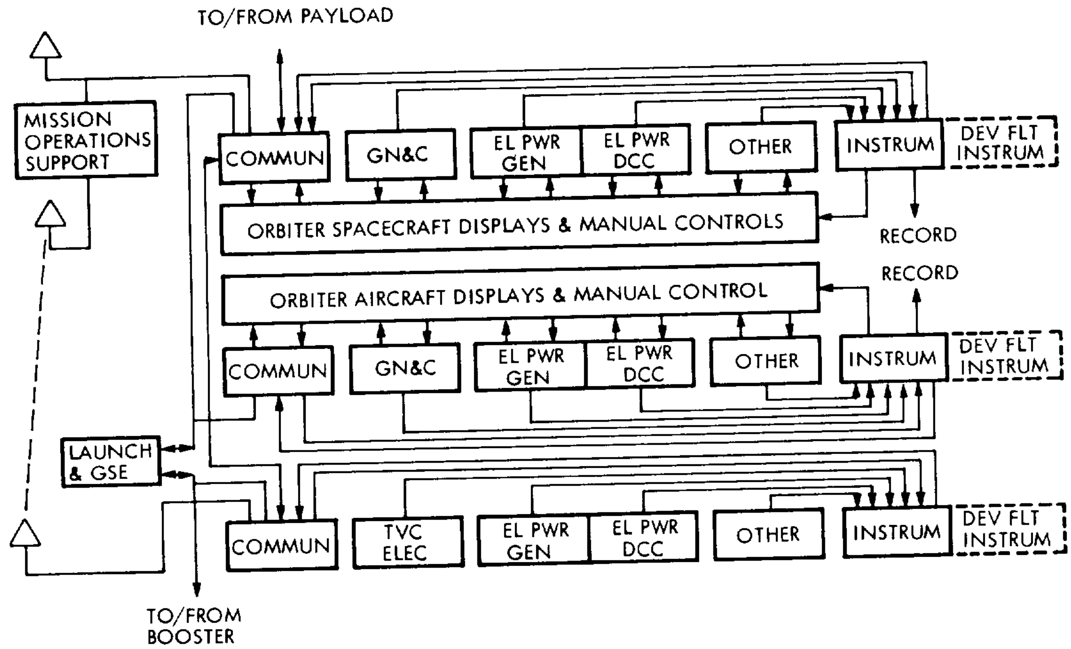
SUMMARY OF BOOSTER STUDIES

MARK I/MARK II SUBSYSTEM STATUS

- AVIONICS
- TPS
- STRUCTURE (ALUMINUM VS TITANIUM)

ALTERNATE A

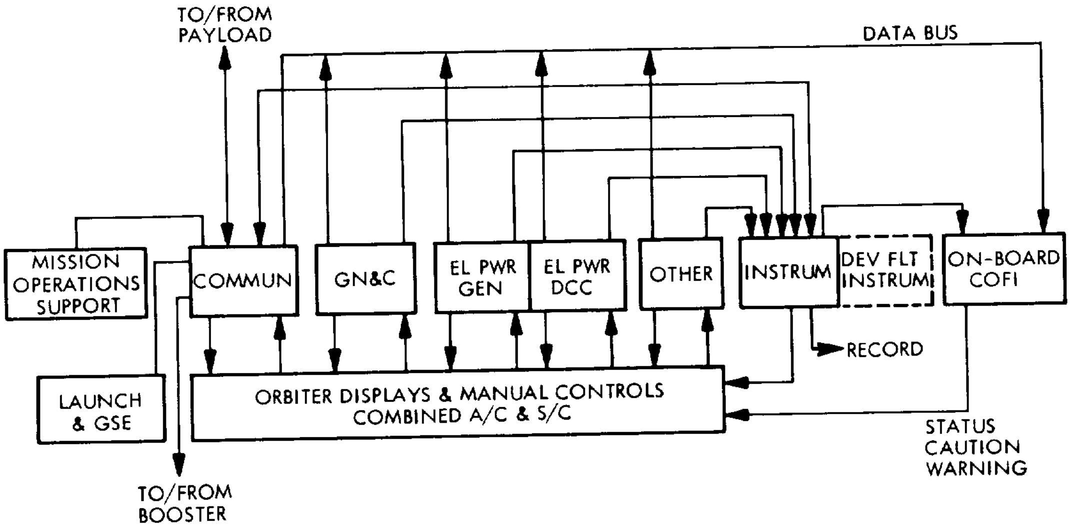
System Alternate A provides separate aircraft and spacecraft subsystems which are dedicated and hardwired. Displays and controls for aircraft and spacecraft functions are provided at completely separate stations. No provision for onboard checkout and fault isolation means that extensive mission operations support from ground facilities is required.


TO/FROM PAYLOAD
MISSION OPERATIONS SUPPORT
COMMUN
GN&C
EL PWR GEN
EL PWR DCC
OTHER
INSTRUM
DEV FLT INSTRUM
ORBITER SPACECRAFT DISPLAYS & MANUAL CONTROLS
RECORD
ORBITER AIRCRAFT DISPLAYS & MANUAL CONTROL
RECORD
COMMUN
GN&C
EL PWR GEN
EL PWR DCC
OTHER
INSTRUM
DEV FLT INSTRUM
LAUNCH & GSE
COMMUN
TVC ELEC
EL PWR GEN
EL PWR DCC
OTHER
INSTRUM
DEV FLT INSTRUM
TO/FROM BOOSTER

D05648

ALTERNATE B

System Alternate B combines aircraft and spacecraft dedicated and hardwired subsystems into one set, eliminating overlapping functional equipment. Displays and manual controls for aircraft and spacecraft functions are combined and intermingled at pilot and copilot stations. An onboard checkout and fault isolation system incorporating a data bus for equipment test access provides status, caution, and warning information to the crew. Dependence on mission operations support from the ground is reduced.

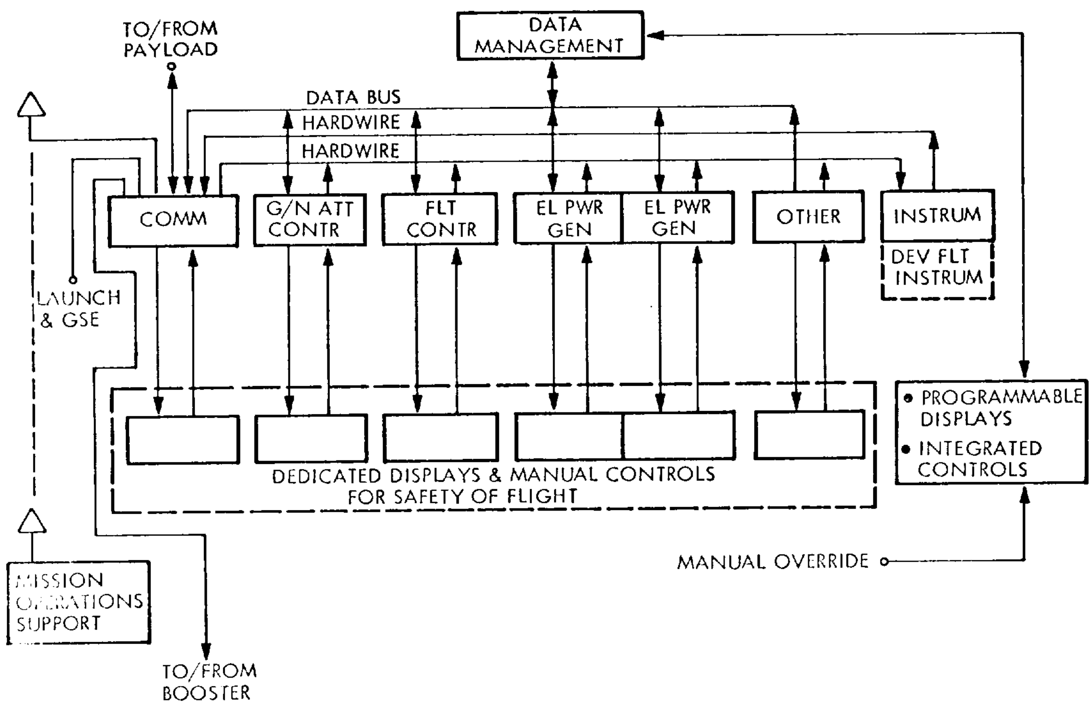

TO/FROM
PAYLOAD
DATA BUS
MISSION
OPERATIONS
SUPPORT
COMMUN
GN&C
EL PWR
GEN
EL PWR
DCC
OTHER
INSTRUM
DEV FLT
INSTRUM
ON-BOARD
COFI
RECORD
LAUNCH
& GSE
ORBITER DISPLAYS & MANUAL CONTROLS
COMBINED A/C & S/C
STATUS
CAUTION
WARNING
TO/FROM
BOOSTER

MARK I ORBITER AVIONICS BASELINE RECOMMENDATION

Avionics equipment already developed for aircraft and spacecraft (e.g., L-1011, C-5A, S-3A, Apollo, Agena) is used throughout the baseline. Equipment modifications are relatively minor. Environmental protection of aircraft-type equipment will be provided. All equipment required for safe return is dedicated and hardwired.

The S-3A Data Management System provides improved on-board checkout and fault isolation, and reduced dependence on mission support from ground stations. The result is a decrease in program cost. The S-3A programmable displays and integrated control panels, together with the Data Management computer, permit the crew to access any information available to the computer.

This Baseline Avionics System provides flexibility for growth to an expanded Mark II capability. The functions assigned to the Data Management computer may be increased to include subsystem computations, with either backup or primary responsibility as desired. Increased vehicle autonomy through on-board mission planning and on-board mission operations support can reduce operations cost.


TO/FROM
PAYLOAD
DATA
MANAGEMENT
DATA BUS
HARDWIRE
HARDWIRE
COMM
G/N ATT
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EL PWR
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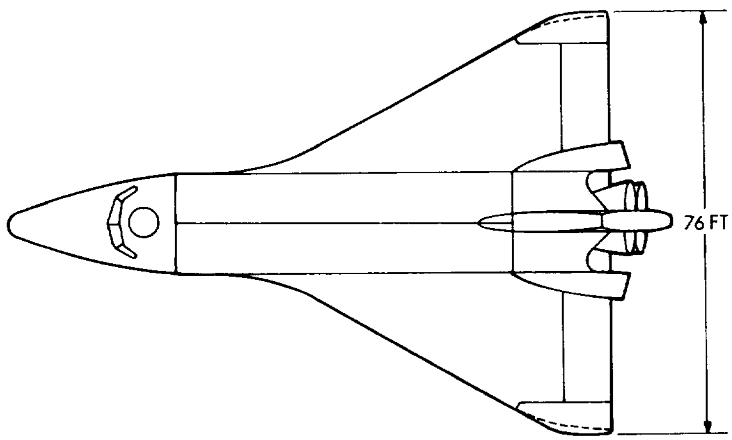
EL PWR
GEN
OTHER
INSTRUM
DEV FLT
INSTRUM
LAUNCH
& GSE
PROGRAMMABLE
DISPLAYS
INTEGRATED
CONTROLS
DEDICATED DISPLAYS & MANUAL CONTROLS
FOR SAFETY OF FLIGHT
MANUAL OVERRIDE
SUPPORT
TO/FROM
BOOSTER

D05683

LMSC MODIFIED 040A ORBITER MARK I - MARK II

Shown here is a three view drawing of the current Lockheed 040A orbiter. The large aileron and rudder areas preclude the use of a mannually powered cable flight control system. In addition to power boost, a mechanical mixer box would be required to compensate for the complex roll-yaw coupling. The high weight of a cable system, its complexity, and the fact that the mixer box would require a two-year development time all favor the use of a stability augmented, fly-by-wire flight control system with appropriate redundancy.

LMSC MODIFIED 040A ORBITER MARK I – MARK II

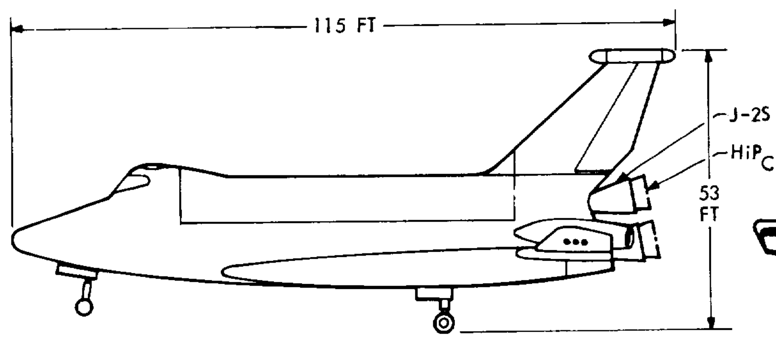


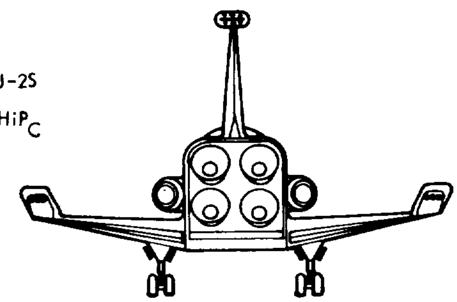
AERODYNAMIC DATA: (AREAS, FT^2)

PLANFORM	3,630
WING (THEOR)	3,180
AILERON	448
WING, WETTED	4,038
VERTICAL TAIL	350
RUDDER	146
VERTICAL, WETTED	746
FUSELAGE, WETTED	6,431

PROPULSION:

MAIN ROCKET ENG.	QTY 4
RCS THRUSTERS	QTY 34
OMPS ENGINES	QTY 2
ABES GE F101/F12 A3	QTY 2





D05598

BASELINE ORBITER AVIONICS

The chart opposite shows the growth of the orbiter avionics beginning with (1) the first horizontal test flight with its complement of aircraft equipment, (2) the vertical test flight with the addition of spacecraft-type equipment, (3) the Mark I operational flight configuration with inertial update and rendezvous capability, (4) the elimination of PCM telemetry, and (5) the operational Mark II configuration with its full complement of avionics, including autopilot/autoland capability.

At the bottom of the chart, the nonrecurring and recurring costs of the Mark I orbiter baseline avionics are shown. The breakdown of these costs is as follows:

<u>Nonrecurring</u>

1. Design and development	$136.7M
2. Hardware and test	56.5M
3. Tooling	1.0M
4. Fab and assembly of FTV-1	7.8M
5. Fab and assembly of FTV-2	17.4M
Total	$219.4M

<u>Recurring</u>

1. Refurbish FTV-1 and FTV-2	$ 11.8M
2. Engineering sustaining	51.3M
3. Vehicle operation (launch, flight, spares)	31.0M
Total	$ 94.1M

LOCKHEED

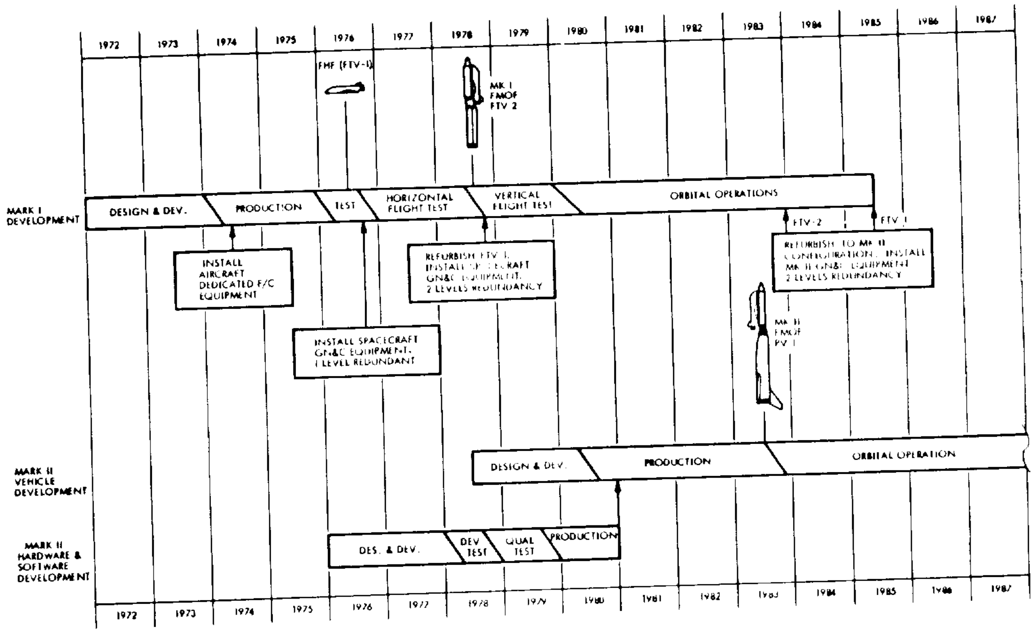
BASELINE ORBITER AVIONICS

MARK I			MARK II
HORIZONTAL TEST FLIGHT	VERTICAL TEST FLIGHT	OPERATIONAL FLIGHTS	
• GN&C • AIRCRAFT "FBW" PILOT CONTROLS • STABILITY AUGMENTATION • AUTOMATIC THROTTLE • FLIGHT DIRECTOR • TACAN	• GN&C SAME AS HORIZONTAL TEST FLIGHT (HTF) EXCEPT ADD • ILS • INERTIAL NAVIGATION • DIGITAL COMPUTER • TVC • ACPS	• GN&C SAME AS VERTICAL TEST FLIGHT (VTF) EXCEPT ADD • INERTIAL UPDATE • RENDEZVOUS	• GN&C • AIRCRAFT "FBW" PILOT CONTROLS • STABILITY AUGMENTATION • AUTOMATIC THROTTLE • FLIGHT DIRECTOR • TACAN • ILS • INERTIAL NAVIGATION (PLATFORM) • GN&C DIGITAL COMPUTER • TVC • ACPS • INERTIAL UPDATE • RENDEZVOUS • AUTOPILOT/AUTOLAND
• DISPLAYS & CONTROLS • AIRCRAFT FLIGHT CONTROLS	• DISPLAYS & CONTROLS SAME AS HTF EXCEPT ADD • S-BAND COMM & CONTROLS • GN & SPACECRAFT ATT CONTROLS • OMPS • ACPS • TRANSLATION CONTROL • FUEL CELL	• DISPLAYS & CONTROLS SAME AS VTF	• DISPLAYS & CONTROLS • AIRCRAFT FLIGHT CONTROLS • S-BAND COMMUNICATIONS & CONTROLS • GN & SPACECRAFT ATTITUDE CONTROLS • OMPS • ACPS • TRANSLATION CONTROL • FUEL CELL
• COMMUNICATIONS & CONTROLS • AIR TRAFFIC CONTROLS • S-BAND, UHF, VHF	• COMMUNICATIONS & CONTROLS SAME AS HTF	• COMMUNICATIONS & CONTROLS SAME AS VTF	• COMMUNICATIONS & CONTROLS • AIR TRAFFIC CONTROL • S-BAND, UHF, VHF
• DATA MANAGEMENT	• DATA MANAGEMENT • DIGITAL COMPUTER • DRUM & MAG TAPE MEMORY STORAGE • LIMITED ON-BOARD CHECKOUT, FAULT ISOLATION, REDUNDANCY MGT	• DATA MANAGEMENT SAME AS VTF	• DATA MANAGEMENT • DIGITAL COMPUTER • DRUM & MAG TAPE MEMORY STORAGE • ON-BOARD CHECKOUT, FAULT ISOLATION, REDUNDANCY MGT
• ELECTRICAL • BATTERY (EMERGENCY) • INVERTERS, GENERATORS • TRANSFORMER RECTIFIERS	• ELECTRICAL SAME AS HFT EXCEPT ADD • FUEL CELLS	• ELECTRICAL SAME AS VFT	• ELECTRICAL POWER • BATTERY (EMERGENCY) • INVERTERS GENERATORS • TRANSFORMER RECTIFIERS • FUEL CELLS
• INSTRUMENTATION • PCM TELEMETRY • CAMERA (FILM)	• INSTRUMENTATION SAME AS HFT	• INSTRUMENTATION	• INSTRUMENTATION
MARK I ORBITER BASELINE AVIONICS COST NON-RECURRING - \$219 M RECURRING - \$ 94 M		MARK II ORBITER BASELINE AVIONICS COST 40 - 50% COST GROWTH OVER MARK I	

The chart opposite shows the phasing of Mark I and Mark II Guidance, Navigation, and Control equipment starting with the aircraft-dedicated equipment required for the first horizontal flight test (FTV-1) in mid-1976. The first spacecraft-type GN&C equipment will be required in late 1976 for installation in the first vertical flight test vehicle (FTV-2). The level of redundancy for the flight test will be one above the minimum required for crew safety. For the operational Mark I, GN&C equipment will be required in late 1978. The level of redundancy will be two levels above the minimum required for crew safety or one level above the minimum required for mission success, whichever is greater.

For Mark II, the first group of GN&C equipment will be needed in January of 1981. FTV-1 will be refurbished to the Mark I operational configuration in late 1981 and refurbished again to the Mark II configuration in mid-1985. FTV-2 will be refurbished to the Mark II configuration the first part of 1984.

GN&C MARK I/MARK II PHASE-IN



ALTERNATE AVIONIC SYSTEM STUDY

(Mark I Orbiter Avionics System Cost Summary)

Included as nonrecurring costs are those costs associated with design and development, software, computers, 1.5 sets of ground test hardware, ground test costs, mockups, simulations, qualification tests, ground support equipment and ground support costs, tooling and FTV-1 and FTV-2 costs for each avionics subsystem.

The recurring costs include hardware and labor to refurbish flight test vehicles to operational status, plus computer costs, and recurring engineering support.

The operations costs include repair and replacement of avionics hardware during Mark I operations, two sets of spare equipment, refurbishment of operational vehicles, and vehicle operation costs during the Mark I program.

MARK I ORBITER AVIONICS SYSTEM COSTS SUMMARY

	BASELINE CONFIGURATION	ALTERNATE A	ALTERNATE B
NONRECURRING	$219.4M	176.7	210.3
RECURRING	11.8M	8.8	8.9
OPERATIONS	82.3M	64.8	68.8
TOTALS	313.5M	250.3	287.9

CONFIGURATION IMPACT

The increased vehicle program costs attendant with providing greater autonomy within the orbiter vehicle are more than offset by the savings from reduced ground support.

Net reductions in program costs up to mid 1985 amount to $22.3M for autonomous capability provided by S-3A technology over C-5A technology and $44.7M over commercial aircraft technology.

Increased autonomy through Mark II in-service growth capability inherent in the Lockheed baseline can reduce Space Shuttle support costs by more than $1B over the program life.

CONFIGURATION IMPACT ($M)

	MARK I			TOTAL OF MARK I AND II		
	A	B	C	A	B	C
MAINT AND LCC	248.6	174.0	149.2	1192.1	834.5	715.3
MCC	77.1	77.1	54.0	337.0	337.0	53.3
REMOTE SITES	5.4	3.5	3.5	15.7	10.4	3.5
TOTALS	331.1	254.6	206.7	1544.8	1181.9	782.9
VEHICLE PROGRAM COSTS	233.8	287.9	313.5			
Δ IMPACT ON SUPPORT COSTS	0	-76.5	-124.4		-362.9	-761.9
Δ VEHICLE COSTS	0	+54.1	+79.7			
NET SAVINGS:						
B AND C OVER A	0	-22.4	-44.7			
C OVER B	-	0	22.3			

DELTA IMPACT FOR FVF - UNMANNED

The unmanned vehicle will require automatic control and sequencing of all on-board subsystems and equipments. The list of time-critical spacecraft functions which are automated in the baseline avionics system will be expanded to include aircraft functions. Software requirements are increased for the orbiter, LCC, MCC, and the MSFN.

Ground controllers and chase plane controllers provide a backup mode to the on-board automatic systems. Extended horizontal flight testing for demonstration of equipment will be required.

DELTA IMPACT FOR FVF – UNMANNED

COST IMPACT

$12.6M	SOFTWARE
7.5M	ORBITER EQUIPMENT
3.5M	ADDITIONAL HORIZONTAL FLIGHT TESTS
5.9M	GROUND SUPPORT & INSTALLATIONS
$29.5M	TOTAL

PROGRAMMATIC IMPACT

- INSTALLED AUTOMATIC SYSTEMS WILL PROVIDE EQUIVALENT OF MARK II CAPABILITY AT EARLIER DATE. THIS MEANS EARLIER PROGRAM COMMITMENT TO MARK II SYSTEM CONFIGURATION

- INCREASED PROGRAM RISK WITHOUT MAN ON-BOARD AS DECISION – MAKER. MANUAL CONTROL MODES AND MANUAL OVER-RIDE OF AUTOMATIC SYSTEMS ARE LACKING ON-BOARD

- IF INTACT VEHICLE RECOVERY IS DESIRED, THE DELTA IMPACT FOR FIRST VERTICAL FLIGHT – UNMANNED IS NEGATED

AGENDA

INTRODUCTION

ORBITER & TANKS

- H VS HO
- TANDEM VS PARALLEL
- INTERSTAGE CONCEPTS
- TANK SEPARATION & DISPOSAL
- TANK DESIGN/COST

MARK I/MARK II ENGINE BASELINE

- J-2 OR J-2S $\longrightarrow$ HiP$_C$
- J-2 OR J-2S FOR MK I/MK II
- HiP$_C$ FOR MK I/MK II
- ENGINE PROGRAM RECOMMENDATION

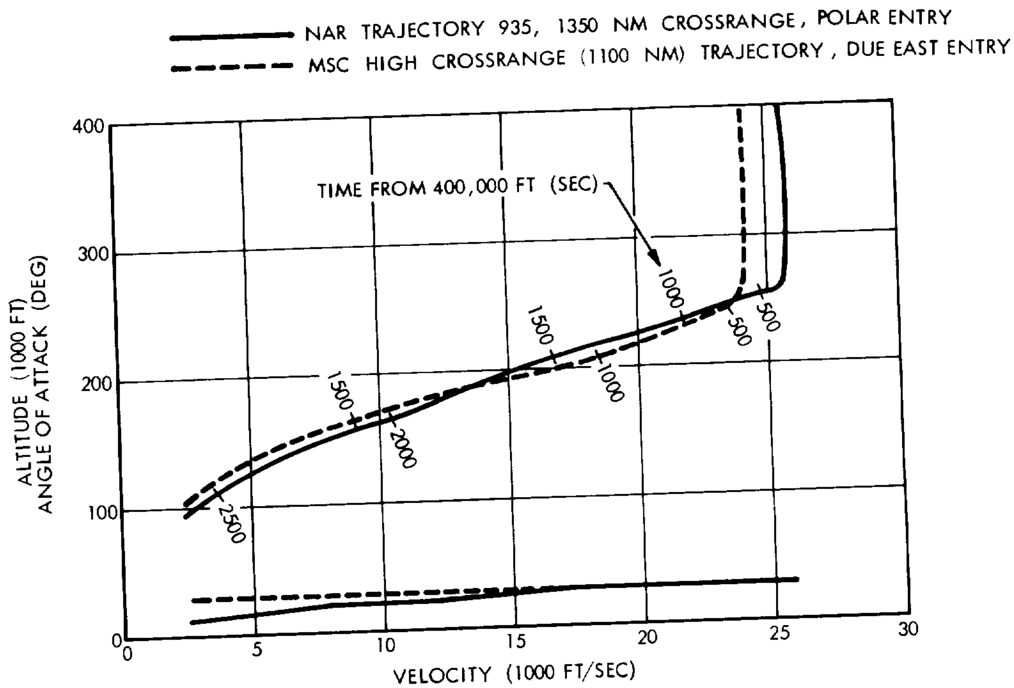
SUMMARY OF BOOSTER STUDIES

MARK I/MARK II SUBSYSTEM STATUS

- AVIONICS
- $\boxed{\text{TPS}}$
- STRUCTURE (ALUMINUM VS TITANIUM)

ENTRY TRAJECTORY COMPARISON – O4OA ORBITER

Aerothermodynamic analyses and TPS sizing for the O4OA Orbiter were accomplished using two trajectories. The first was an NAR Phase B entry trajectory selected by MSC for preliminary analysis. The second was generated by MSC. The NAR trajectory is based upon entry from a polar orbit (turning east) and results in an aerodynamic crossrange of 1350 nm. The MSC trajectory is based upon due east entry and, as shown in the chart opposite, results in higher heating at high velocities. Total entry times from 400,000 ft to touchdown are 3350 and 2650 sec for the NAR and MSC trajectories, respectively.


NAR TRAJECTORY 935, 1350 NM CROSSRANGE, POLAR ENTRY
MSC HIGH CROSSRANGE (1100 NM) TRAJECTORY, DUE EAST ENTRY
TIME FROM 400,000 FT (SEC)
ALTITUDE (1000 FT)
ANGLE OF ATTACK (DEG)
VELOCITY (1000 FT/SEC)
400
300
200
100
0
5
10
15
20
25
30
500
1000
1500
2000
2500

D05593

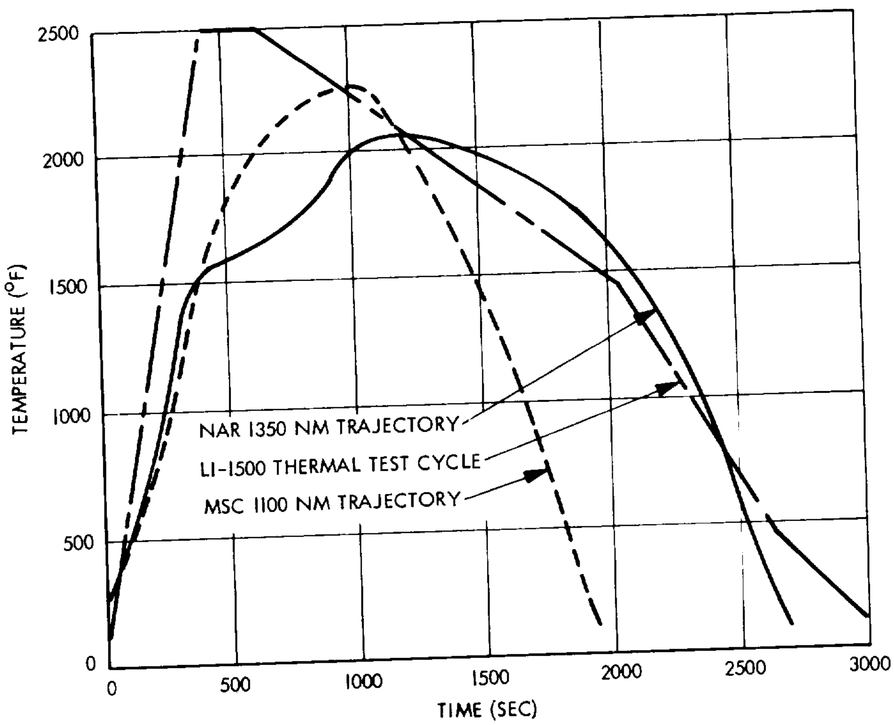
O4OA ORBITER SURFACE TEMPERATURE HISTORY COMPARISON

Temperature-time histories during entry are shown for the O4OA Orbiter lower-surface centerline at 50 percent vehicle length. These are radiation equilibrium temperatures for a surface emittance of 0.8. Delta-wing laminar wind tunnel data (from tests at NASA ARC, LaRC, and AEDC) were used in determining the O4OA entry thermal environment. Boundary layer transition and turbulent heating were accounted for using the methodology recommended by the NASA Thermal Panel.

As shown, peak entry temperature at this location is 2250° and 2050°F for the MSC and NAR entry trajectories, respectively. Also plotted is the thermal test cycle being used for reusable surface insulator (LI-1500) material development. The test pulse is conservative compared to the entry predictions in terms of both peak temperature level and heat load.

040A ORBITER SURFACE TEMPERATURE HISTORY COMPARISON

LOWER SURFACE CENTERLINE, X/L = 0.5

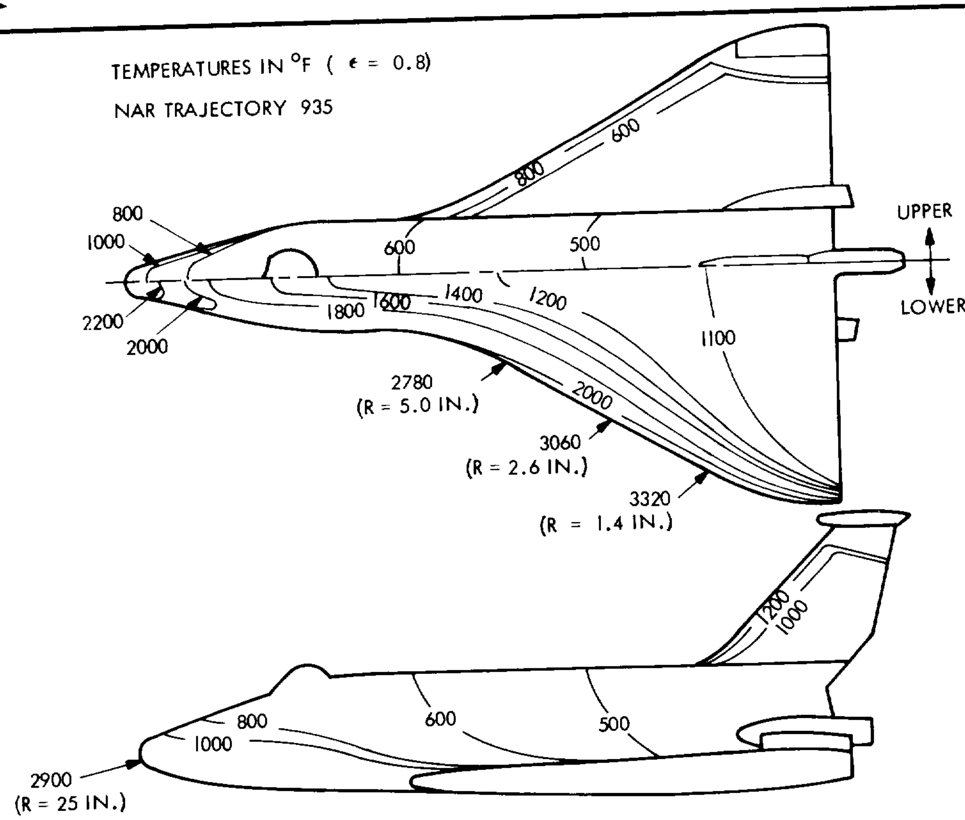


D05831

O4OA PEAK TEMPERATURES BASED ON ALL-LAMINAR FLOW

Peak temperature isotherms are shown for the O4OA Orbiter based on the NAR entry trajectory and direct extrapolation of the wind tunnel test data to flight conditions. Since the test data were obtained with a laminar boundary layer, this extrapolation is equivalent to assuming an all-laminar flow during entry. As shown, the peak temperature occurs outboard on the wing leading edge. On most of the lower surface, peak temperatures are between 1100° and $1800^{\circ}F$.

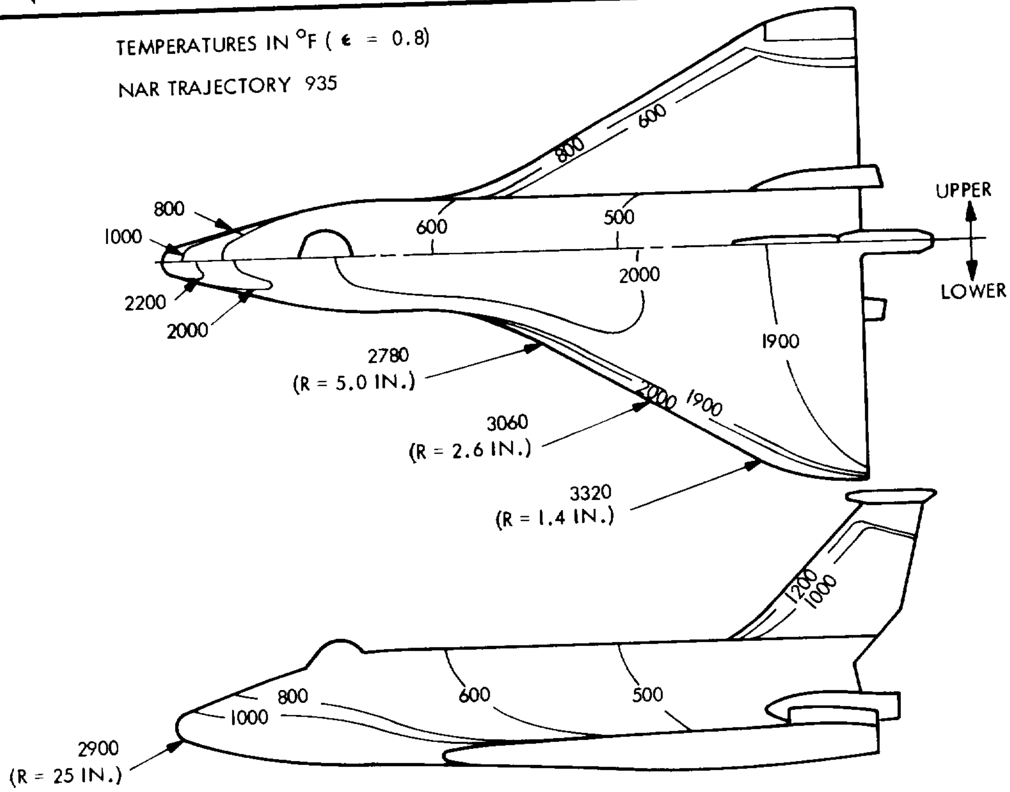
040A PEAK TEMPERATURES BASED ON ALL-LAMINAR FLOW



DO5844

If, in the prediction of the entry thermal environment, allowance is made for boundary layer transition and turbulent heating, a significant increase in temperature level will result in affected surface areas. Peak temperatures, shown on the chart opposite, are based upon transition and turbulent heating criteria recommended by the NASA Thermal Panel. Peak temperatures on the lower surface are between 1900° and 2100°F and are generally 600° to 700°F higher than if predicted on the basis of laminar flow only. Temperatures are unchanged in those areas where the flow is predicted to be laminar at peak entry heating.

040A PEAK TEMPERATURES INCLUDING TRANSITION



DO5857

COMPARISON OF MARK I AND MARK II TPS – O4OA ORBITER

Mark I and Mark II TPS thicknesses and weights are tabulated for various O4OA Orbiter locations based on the NAR entry trajectory. Thicknesses are average values for the surface areas shown. By coincidence, the total TPS weights for the two concepts are identical, although the weights at specific vehicle locations vary. For example, on the body lower surface, the Mark I ablative TPS is lighter than the Mark II reusable surface insulation as a result of increased efficiency of the ablators in high heating areas. On leeward surfaces, the ablator is thicker than the reusable insulation, since it performs purely as an insulator.

COMPARISON OF MARK I AND II TPS - 040A ORBITER

NAR TRAJECTORY 935

LOCATION	SURFACE AREA (FT^2)	MARK I*		MARK II**	
		THICKNESS (IN.)	WEIGHT (LB.)	THICKNESS (IN.)	WEIGHT (LB.)
BODY LOWER SURFACE	1,647	2.30	5332	2.50	6176
BODY UPPER SURFACE	4,757	1.10	7912	0.75	6917
WING LOWER SURFACE	1,956	2.20	6076	2.40	7078
WING UPPER SURFACE	1,956	0.70	2227	0.60	2459
TAIL (SIDES)	722	1.50	1580	0.85	1146
NOSE CAP	27	2.75	201	---	95
WING LEADING EDGE	126	2.75	937	---	441
TAIL LEADING EDGE	24	2.00	131	---	84
TOTAL	11,215		24,396		24,396

*NOSE CAP & LEADING EDGE TPS IS 30 LB/FT^3 ESA 3560 ABLATOR; REMAINDER IS 15 LB/FT^3 SLA 561

**NOSE CAP & LEADING EDGE TPS IS CARBON-CARBON; REMAINDER IS 15 LB/FT^3 LI-1500 (RSI)

DO5860

flg

Replacement of an ablative thermal protection system after each flight results in prohibitive costs when considering its use over the total operations program of 445 flights. Comparison of total program costs is shown for the SLA 561 ablator TPS versus the LI-1500 Reusable Surface Insulation TPS based on the following considerations:

SLA 561 Ablator (445 Flights)

\$ 65M	Development
1070M	Operations at \$2.4 M/flight (\$180/ft^2 x 11,000 ft^2 x 1.20)
\$1135M	

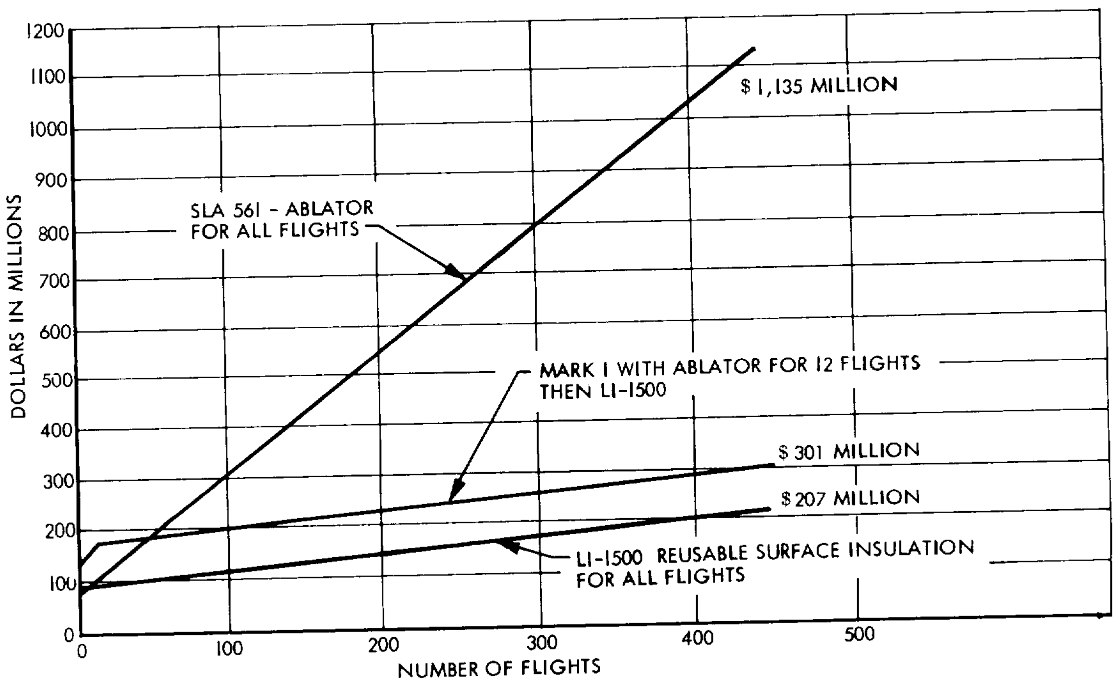
LI-1500 RSI (445 Flights)

\$ 76M	Development
26M	6 Vehicles at \$4.3M/flight (\$320/ft^2 x 11,000 ft^2 x 1.20)
105M	Inspection (0.1 hr/ft^2) and refurbishment at 5 percent per flight
\$ 207M	

Ablator (12 Flights); LI-1500 (445 Flights)

\$ 65M	Development
29M	12 Flights
207M	445 Flights
\$ 301M	

TPS COST COMPARISON



DO5843(1)

AGENDA

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SUMMARY OF BOOSTER STUDIES

MARK I/MARK II SUBSYSTEM STATUS

- AVIONICS
- TPS
- $\boxed{\text{STRUCTURE (ALUMINUM VS TITANIUM)}}$

D05786 (14)

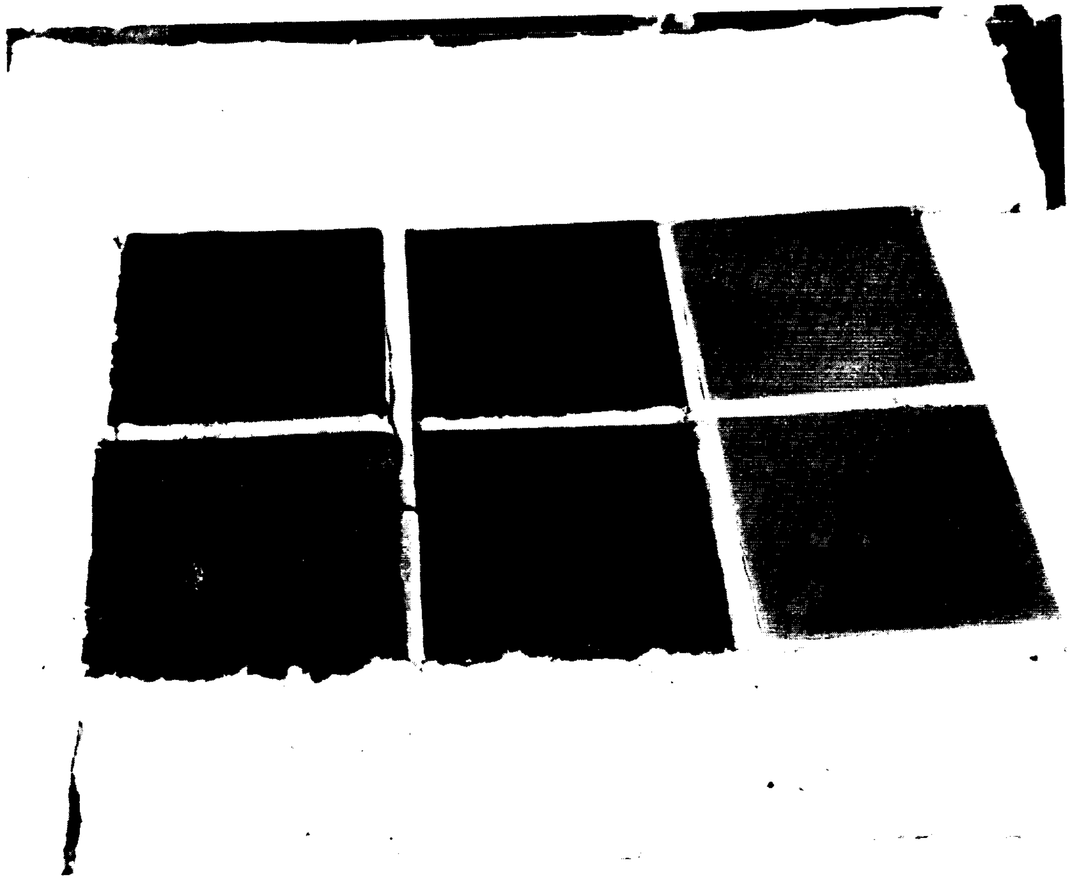
LI-1500 SPECIMENS PRIOR TO TEST

The photo opposite depicts six 4 in. x 4 in. x 2.45 in. LI-1500 specimens with different surface coatings prior to radiant heat exposure. Specimens on the left were coated with an integral coating with a silicon carbide emittance agent, while the specimens on the right were coated with a new addon-type borosilicate coating with a silicon carbide emittance agent. Center specimens were coated with a borosilicate coating with a chrome oxide emittance agent. These specimens were instrumented with 18 surface and indepth thermocouples to record temperatures and verify the repeatability of the thermal properties.

LI-1500 SPECIMENS AFTER TEST

The photo opposite depicts the LI-1500 specimens after exposure to 97 cycles of the 1100-nm crossrange reentry temperature pulse. The time at the peak temperature of 2500°F was 2-1/2 minutes of the total heating pulse of 50 minutes.

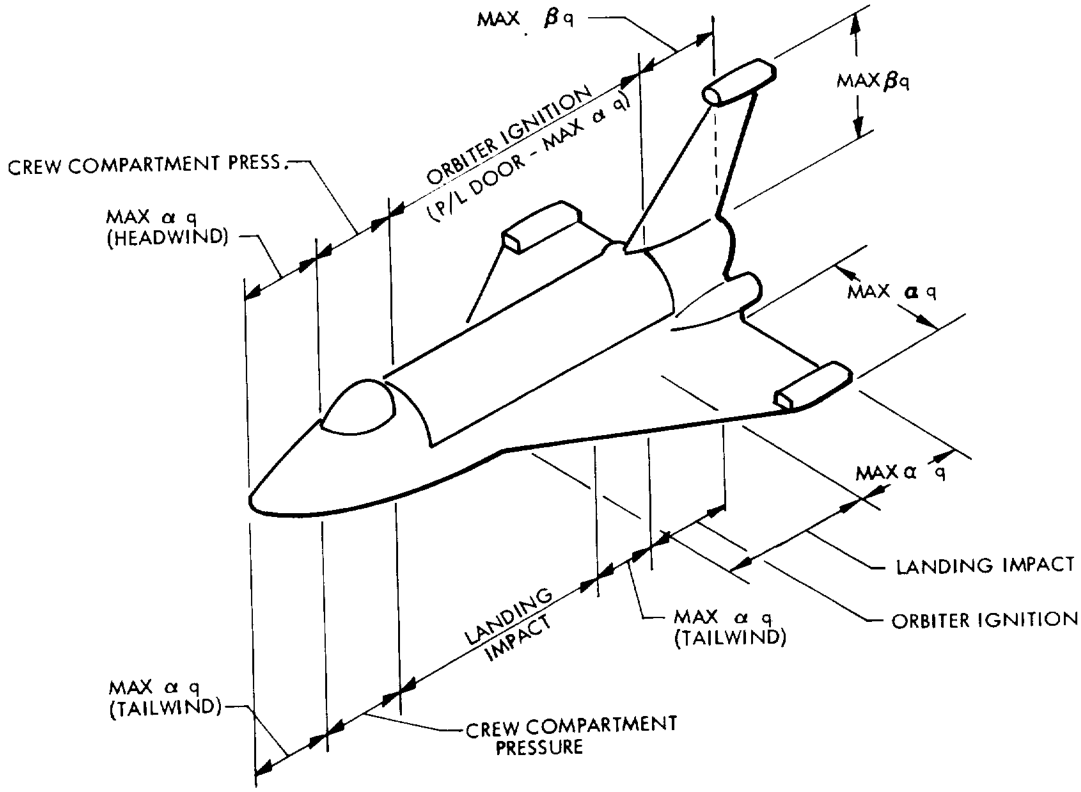
Successful results of the test are best illustrated by the excellent appearance of the new borosilicate/silicon carbide coating on the two specimens on the right of the photo. The repeatability of the indepth temperature measurements over the 97 cycles indicates the stability of the thermal properties and applicability of LI-1500 for the Space Shuttle Thermal Protection System.



Design loads for the maximum in-flight wind conditions were calculated on the basis of a headwind and tailwind αq of 3000 deg-psf. A βq of 4500 deg-psf was assumed for a sidewind and the maximum dynamic pressure was 650 psf. Four engines of 265,000 lb thrust per engine, canted 11.5 deg to point the thrust vector through the composite cg, constituted the basis of the orbiter ignition condition.

At landing, a 2.5g impact was assumed, 1.0g lift on the wing and 1.5g through the main gear. A 2.5g and -1.0g symmetrical maneuver for post-entry flight and a 3g end-of-boost condition were found to be less critical than the above conditions.

DESIGN LOAD DIAGRAM – ORBITER

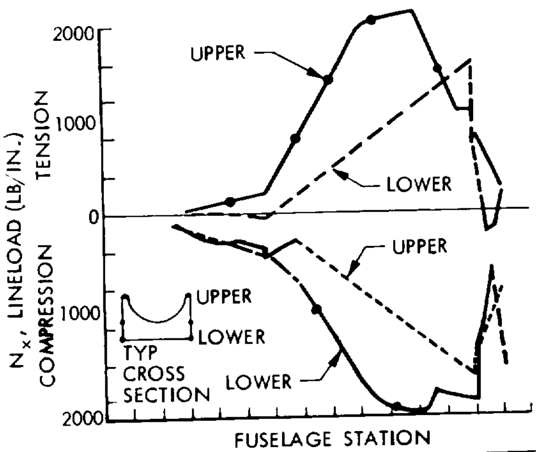


FUSELAGE PANEL DESIGN REQUIREMENTS

Orbiter fuselage line load and section properties were obtained using the FAST computer code. Line loads, based on 6 to 13 nodal points per half-fuselage cross section, were obtained at 9 fuselage stations. Five loading conditions were considered.

The structural model assumes a nonload-carrying payload door with major longerons located at the payload door sill and at the wing-fuselage intersection (i.e., upper and lower caps of the wing-root rib).

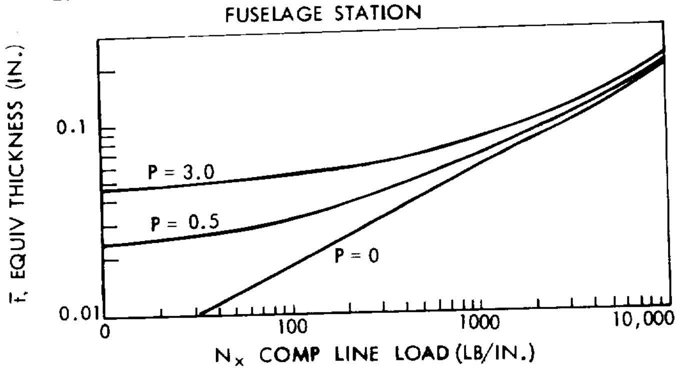
Using the zee-stiffened panel design chart, fuselage line loads and aerodynamic pressures were matched at selected nodal points to determine panel sizes and equivalent thicknesses. The panel sizes and equivalent thicknesses were used to determine initial fuselage finite element stiffnesses for the REXBAT finite element orbiter model. Panel design bending moments due to airloads were 'magnified' to account for beam column effects (increased bending due to interaction of panel deflections and the in-plane compressive line loads).



ULTIMATE LINE LOAD ENVELOPE

CONDITIONS: MAX αq (HEADWIND), MAX αq (TAILWIND), BOOSTER BURNOUT, ORBITER IGNITION } FS = 1.4

LANDING IMPACT, FS = 1.5



EQUIVALENT PANEL THICKNESS

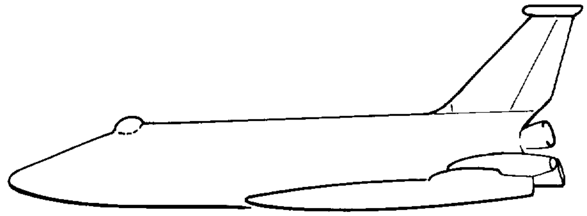
VS

LINE LOAD AND PRESSURE

ORBITER THICKNESS REQUIREMENTS

Preliminary effective skin gages are based on analysis using preliminary design load conditions summarized in the previous chart titled Design Load Diagram – Orbiter.

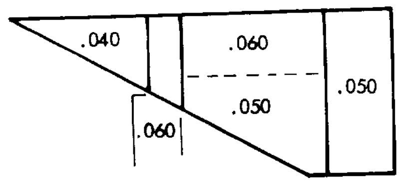
ORBITER THICKNESS REQUIREMENTS



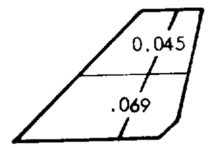
FUSELAGE PANEL EQUIV. THICK

STA	UPPER SURFACE	LOWER SURFACE
300	0.064	0.058
400	0.068	0.070
500	0.068	0.084
600	0.072	0.058
800	0.046	0.074
1016	0.058	0.086
1200	0.070	0.092
1314	0.071	0.082
1400	0.052	0.073

WING PANEL EQUIV. THICK



FIN PANEL EQUIV. THICK.



DO5908

AIRFRAME STRUCTURE/TPS COSTS

- Costs are shown as Δcosts relative to the all-aluminum version with LI-1500 RSI bonded directly to primary structure
- Basic Cost Estimating Relationships (CERs) are:

  Total Program Cost: DDT&E + Recurring Production + Recurring Operations

  DDT&E; Development Cost + 4 x First Unit Cost

  Recurring Production: 3 x First Unit Cost

  Recurring Operations: Based on 445 Flights

AIRFRAME STRUCTURE / TPS COSTS*

DESIGN CONFIGURATION	STRUCTURE & TPS (1)			TOTAL SYSTEM (2)			
	TPS	STRUC.	SUM	DDT&E	RECURR. PROD.	RECURR. OPS.	SUM
SYSTEM 5B	40.4	101.4	141.8	132	66	11	209
ALL-ALUMINUM	0	0	0	0	0	0	0
MAXIMUM TITANIUM	-9.8	286.0	276.2	271	130	30	431

*DIFFERENCE IN COSTS ($M)

(1) COSTS OF ORBITER STRUCTURE AND TPS ONLY

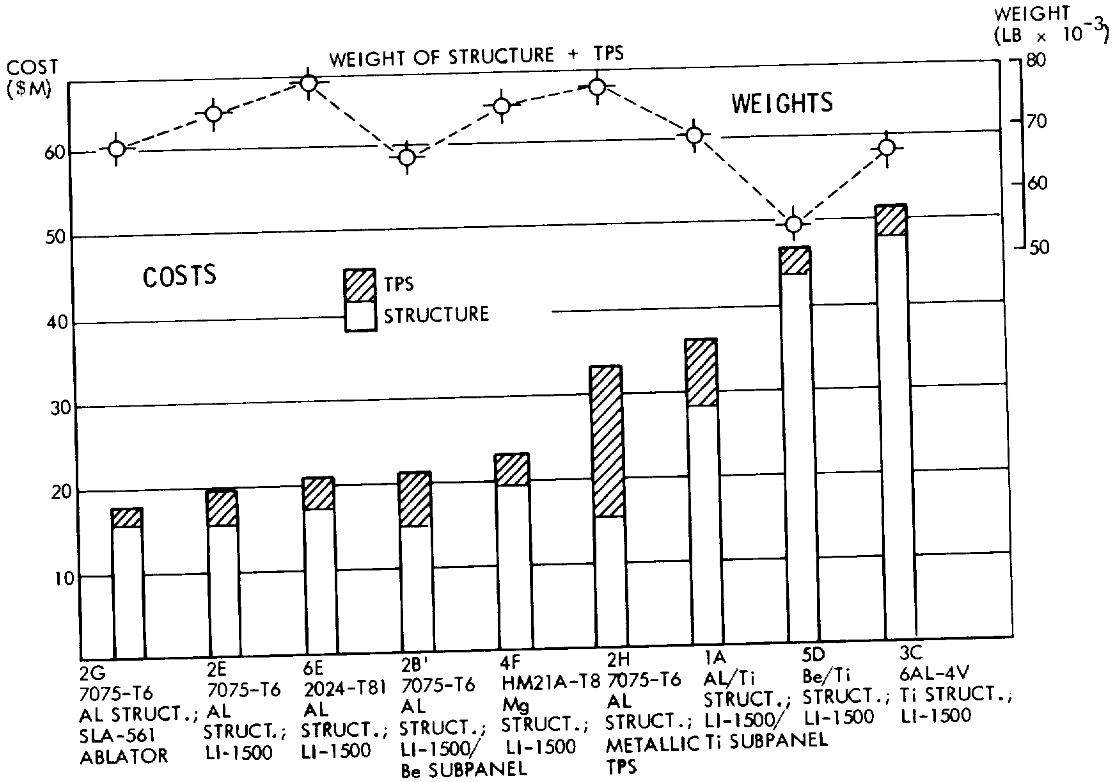
(2) TOTAL SYSTEM CHANGES REFLECT CHANGES IN ORBITER AND BOOSTER DRY WEIGHTS, OLOW, AND GLOW RESULTING FROM STRUCTURE/TPS WEIGHT DIFFERENCES

ORBITER STRUCTURE/TPS FIRST UNIT COST COMPARISON

A large matrix of candidate airframe structural materials and thermal protection systems (TPS) for the Configuration 5B Orbiter has been evaluated for variances in program costs. Primary structure materials considered included 7075-T6 and 2024-T81 aluminum alloys, 6Al-4V titanium, HM21A-T8 magnesium, and cross-rolled beryllium sheet. TPS design configurations included (1) LI-1500 RSI material bonded to titanium or beryllium sub-panels, (2) LI-1500 RSI or SLA-561 ablator material bonded directly to primary structure, and (3) a metallic system using Coated Cb, TD NiCrAl, Rene 41, and titanium as surface panel materials, and titanium for the subpanels carrying aerodynamic pressure loading.

The chart opposite illustrates the variance in costs for both the primary structure and TPS, and also shows the total structure/TPS weight for each design configuration. The ablative TPS design shows the least cost on a single-unit basis but, since it must be replaced after each flight, results in prohibitive costs on a total program basis. The variance in structure costs reflects the large difference in manufacturing complexity existing between aluminum, titanium, and beryllium materials.

ORBITER STRUCTURE/TPS FIRST UNIT COST COMPARISON



D05926

Conclusions. The static/dynamic finite-element computer model permits rapid capability of assessing the structural sizing and stiffness requirements of the orbiter vehicle. Preliminary results show that the orbiter fuselage, wing, and fin line load magnitude is less than 2000 ppi. This range is within the design requirements for typical payload shrouds. The resulting skin gage in aluminum for this range is 0.020 to 0.040 in. with airload pressure taken into account. A comparable design in titanium results in very thin sheet gages but requires greater manufacturing complexity. A weight penalty in titanium will probably result due to minimum-gage constraints. Aluminum is therefore considered to be cost-effective for this specific design application.

Issues. The integrated evaluation of the orbiter structures/TPS system will continue to (1) define the dynamic pressure conditions and vehicle transient response loads with a 6-D loads analysis; (2) establish a baseline static/dynamic finite-element model by which the vehicle loads and dynamic responses can be studied to facilitate refined structural sizing to include thermal loads, stiffness, and optimized weight; additionally, fail-safe and safe-life requirements can be quickly evaluated with the model by inserting failed members and observing the results, both statically and dynamically; and (3) evaluate structural concepts for the baseline orbiter and tanks to establish various alternate design approaches to perform cost-effectiveness evaluations leading to final baseline selections.

CONCLUSIONS

- STATIC/DYNAMIC MODEL FACILITATES SIZING AND STIFFNESS REQMTS
- AIRFRAME LINELOADS < 2000 PPI — TYPICAL SPACECRAFT SHROUD REQMTS
- ALUMINUM SKIN GAGE RANGE: $0.020 \leq t \leq 0.040$
- TITANIUM SKIN GAGE REQMTS INCREASE STRUCTURAL COMPLEXITY FOR CONSTANT WEIGHT
- ALUMINUM DESIGN RESULTED IN MAXIMUM COST-EFFECTIVENESS FOR THE LOAD REQMTS

ISSUE – CONTINUE INTEGRATED EVALUATION OF ORBITER STRUCTURES/TPS SYSTEM

- DEFINE DYNAMIC PRESSURE CONDITIONS WITH 6-D LOADS ANALYSIS
- DEFINE TRANSIENT LOADS RESPONSES
- STATIC/DYNAMIC F/E MODEL CONTINUED DEVELOPMENT TO FACILITATE SIZING, STIFFNESS, AND FAIL-SAFE/SAFE-LIFE REQMTS
- STRUCTURAL CONCEPTS REEVALUATION
- BASELINE SELECTION AND COST EVALUATION

The baseline system consists of the reusable O4OA orbiter, single external LOX/hydrogen tank and flyback reusable S1-C booster. For Mark I operations, the orbiter uses the J-2S engine and phases over to a HiP_c 261,000-lb thrust engine for Mark II. The booster uses the F-1 engine for both Mark I and Mark II operations. Costs for the J-2S and F-1 engines were obtained from MFSC. For these estimates, a 7-1/2 percent fee was subtracted from the original data to make them consistent with all other estimates. Estimates of booster DDT&E and recurring production costs were also obtained from MSFC and are based on Boeing data.

The Mark I orbiter DDT&E estimate of $1461 million includes two flight test vehicles which later become the two Mark I operational vehicles. The $28 million of Mark I recurring production cost is the cost to retrofit these vehicles to Mark I operational status. The $252 million of Mark II recurring production cost includes retrofitting the two Mark I orbiters to the Mark II configuration plus the production of three additional Mark II orbiters. No recurring production cost is shown for Mark I boosters, for these are assumed to be covered as two flight test boosters under the $1156 million of booster DDT&E. The Mark II booster recurring production cost of $346 million includes the cost of retrofitting the two Mark I boosters to the Mark II configuration plus the production of two additional Mark II boosters.

All costs are in 1970 dollars and exclude fee.

BASELINE PROGRAM COSTS ($M)

	MARK I				MARK II			
	DDT&E	REC. PROD.	REC. OPS.	TOTAL	DDT&E	REC. PROD.	REC. OPS	TOTAL
ORBITER	1464	28	329	1821	297	252	231	780
BOOSTER	1156	0	37	1193	104	346	99	549
TANKS	180	0	167	347	0	0	328	328
ENGINE								
ORBITER	74	91		165	384	52	26	462
BOOSTER	36	124		160	0	324		324
FLIGHT TEST	149	0	0	149	0	0	0	0
OPERATIONS	121	0	591	712	52	0	1156	1208
MGMT & INTEG	315	3	115	433	86	67	186	339
PHASE TOTAL	4980				3990			
PROGRAM TOTAL	8970							

PEAK FUNDING: $991M IN FY '76

MARK I PROGRAM: 123 FLIGHTS
MARK II PROGRAM: 322 FLIGHTS
TOTAL 445 FLIGHTS

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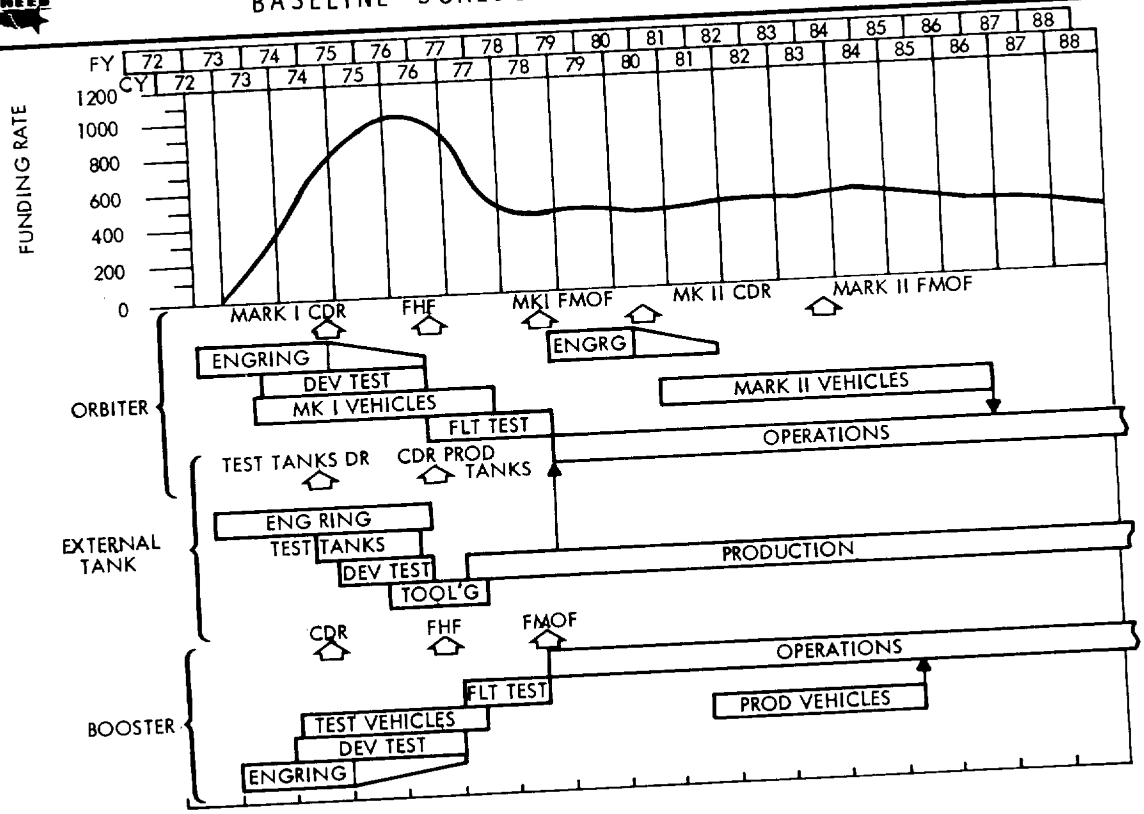
BASELINE SCHEDULE CHARACTERISTICS

A baseline highly condensed schedule is shown for the Mark I/Mark II concurrently developed orbiter and booster. Mark I FMOF is in late 1978, with Mark I FMOF following five years later.

Plotted here are the annual costs in 1970 dollars, peaking in FY 1976 at $991M. The characteristics of this concurrent booster development approach vary from the phased booster approach studied earlier which resulted in a second annual funding peak slightly higher than the initial peak.

Subsequent funding during the Operations phase (i.e., CY 1983 to 1986) could be lowered by pulling the Mark II orbiter back earlier and bringing in the production boosters as required by the mission traffic. This approach would provide a cushion for the buildup of payload costs that begins with the traffic model buildup commencing in FY 1979.

BASELINE SCHEDULE CHARACTERISTICS



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